CDF Study Report

Phobos Sample Return

Phobos Moon of Mars Sample Return Mission

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FRONT COVER

Study logo showing Phobos Sample Return Stack with Mars and Phobos and Deimos in the background

STUDY TEAM

This study was performed in the ESTEC Concurrent Design Facility (CDF) by the following interdisciplinary team:

TEAM LEADER			
AEROTHERMO DYNAMICS		PAYLOAD/ DETECTORS	
GNC		POWER	
COMMUNICATIONS		PROGRAMMATICS/ AIV	
CONFIGURATION		PROPULSION	
COST		RISK	
DATA HANDLING		ROBOTICS	
GS&OPS		STRUCTURES	
MECHANISMS		SYSTEMS	
MISSION ANALYSIS		THERMAL	

Under the responsibility of:

SRE-FMP, Study Manager

SRE-FMP, Study Responsible

With the technical support of

SRE-FMP, System Support

TEC-MSM, Mechanisms (simulations)

SRE-FMP, Observer

SRE-FMP, Observer

With the scientific assistance of:

IKI, Study Scientist

SRE-S, Study Scientist

With the involvement of:

IKI, Technical Support on Sampling chain

IKI, Technical Support on Sampling chain

IKI, Technical Support on Sampling chain

Lavochkin, Technical Support on ERV and PM
Lavochkin, Technical Support on ERV and PM
Lavochkin, Technical Support on ERV and PM

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1 INTRODUCTION

1.1 Background

In 2013 a cooperation agreement was signed between ESA and ROSCOSMOS concerning Cooperation on the Robotic Exploration of Mars and other Bodies in the Solar System

Bilateral discussions concluded that the Russian and ESA plans and views for Mars exploration are quite similar in several aspects, both having Mars Sample Return as long term objective, and both with an interest in a Phobos Sample Return mission as intermediate mission to Mars Sample Return.

A potential cooperation on a sample return mission to Phobos has been agreed as interesting to be jointly investigated, and can be seen as a combination of the Russian mission Boomerang (with an updated definition with respect to Phobos-Grunt) and the proposed ESA-SRE mission Phootprint.

1.2 Scope

This study was an assessment to determine the feasibility of a joint European/Russian Phobos Sample Return Mission. On the ESA side, it was requested by ESA Science Directorate and funded by the General Studies Programme. The Russian contributions were funded by Roscosmos. The study was carried out in the ESA Concurrent Design Facility (CDF) by an interdisciplinary team of specialists from the following organisations:

- IKI (RU)
- Lavochkin (RU)
- ESA-ESTEC (NL)
- ESA-ESOC (DE)
- ESA-ECSAT (UK).

The study was carried out in 8 sessions, starting with a kick-off on the 15th April 2014 and ending with a final presentation on the 20th May 2014.

The scope of the study was to assess the technical feasibility of a joint mission and to create preliminary designs of the various elements, based on a preliminary cooperation scenario agreement between ESA and ROSCOSMOS for mission element sharing (one baseline and one backup solution). The study assessed the technical impact of possible mission options and assessed the technical risk and programmatic aspects of the mission. Specific science and technology goals are detailed in the Mission Objectives chapter of this report.

1.3 Document Structure

The layout of this report of the study results can be seen in the Table of Contents. The Executive Summary chapter provides an overview of the study.

Due to the specific organisation of the study involving ESA, IKI and Lavochkin, the structure of the document has the following particularities:

- The Earth Return Vehicle (ERV) and Propulsion Module (PM) (for baseline configuration) designs are detailed by Lavochkin in stand-alone chapters, 6 and 7, (i.e. the corresponding subsystems description are included in these chapters, while for the Landing Module (LM) subsystems descriptions are addressed later on in the document as part of separate chapters – see below)
- The composite general design is addressed in the system part, chapter 8
- After the system part, the LM (for the baseline configuration) and ERC designs, are described through several chapters addressing the different LM subsystems and the ERC, since these were the focus of the ESA technical team in terms of design definition
- The sampling tool and associated sampling chain concepts are addressed only at the end of the document in two separate chapters (chapters 24 and 25), since two different solutions were proposed by ESA and IKI. However the ESA robotic arm, that could be used for the 2 sampling tool solutions, is described in chapter 12.
- The programmatic parts (programmatics & risks) are provided only from the ESA perspective.

Due to the different document distribution requirements, this document is distributable to the public and therefore no costing information is provided..

2 EXECUTIVE SUMMARY

2.1 Study Flow

Requested by the ESA Science Directorate and funded by the General Studies Programme (GSP), the Phobos Sample Return study was carried out in 8 concurrent sessions starting with a kick-off on the 15th April 2014 and ending with an Internal Final Presentation (IFP) on 20th May 2014. The interdisciplinary team consisted of specialists from IKI (RU), Lavochkin (RU), ESA-ESTEC (NL), ESA-ESOC (DE), and ESA-ECSAT (UK).

2.2 Mission Objectives and Requirements

The Phobos Sample Return CDF study aims at demonstrating the feasibility of a joint ESA-ROSCOSMOS mission studying bulk characteristics of the Martian moons Phobos and Deimos and returning surface samples from Phobos.

The mission is intended to demonstrate and mature technologies required for Mars Sample Return missions as well as to scientifically characterise the Mars moons Phobos and Deimos.

The main mission requirements are summarised in Table 2-1.

Req. ID	Requirement
MI - 010	The mission shall return approximately 100g of loose material from the surface of Phobos
MI - 020	The mission shall perform a series of science measurements of Deimos and Phobos using the payload as defined in chapter 5.
MI - 030	The mission shall be compatible with the science requirements defined in chapter 3.2, and with additional science objectives (following ESA/IKI discussions) defined in chapters 3.2 and 5.
MI - 040	The mission shall be designed for a launch in 2024 as a baseline, with 2026 as back up *C1: any type of transfer identified by mission analyses in this timeframe shall be checked* *C2: the requirement implies that the composite design shall be compliant with both the baseline and backup launch dates* *C3: mission compatibility with 2022 and 2028 launch dates shall also be checked for information*
MI - 050	The mission shall be launched by Proton-M from Baikonur in a direct escape trajectory. *C: the possible interest of an injection in an intermediate Earth orbit followed by a set of manoeuvres performed by the PM for escaping Earth, shall be checked*
MI - 060	The launch windows and transfers characteristics shall be as per chapter 4 of this report.

Req. ID	Requirement
MI - 070	At Mars arrival, the mission shall perform a series of manoeuvres, as per chapter 4 of this report, in order: - First to allow for Deimos characterisation with the science payload, by reaching a QSO around Deimos - Then to reach its operational orbit (QSO) around Phobos *C: in case of non-compliance with the launch mass, fly-by around Deimos instead of QSO may be considered*
MI - 080	The Phobos science characterisation measurements shall be performed from three types of orbits: - A trailing orbit, at the end of the phasing phase, when the spacecraft is on an almost-Phobos-orbit and is getting closer to Phobos - A Quasi Satellite Orbit around Phobos, also called operational orbit, during which a pre-selection of landing sites is performed - Fly-bys orbits over the pre-selected sites for finalising the landing site selection *C: The Deimos observation strategy shall be less exhaustive than for Phobos, and will be defined during the study*

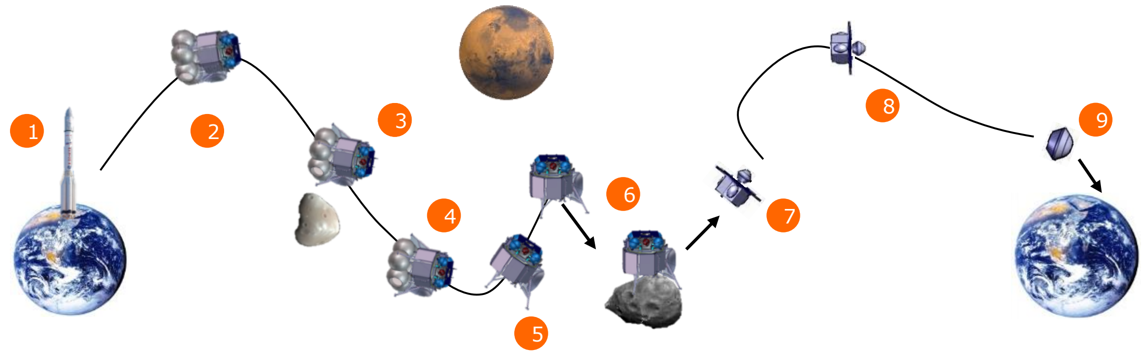
Table 2-1: Main Mission Requirements

2.3 Phobos Sample Return Mission Architecture

The Phobos Sample Return mission consists of four elements: the Propulsion Module (PM), the Landing Module (LM), the Earth Return Vehicle (ERV), and the Earth Re-entry Capsule (ERC). In the baseline cooperation scenario, the PM and ERV are under ROSCOSMOS responsibility while ESA is responsible for the LM and the ERC.

The assembly of the four elements – henceforth called ‘composite’ - is launched with a Proton-M launch vehicle and brought into a direct escape trajectory with the Breeze-M upper stage.

The baseline mission architecture is depicted in Figure 2-1.



1 Launch and Direct Escape with Proton and Breeze-M

2 Transfer Earth-Mars

3 Transfer to Deimos and Deimos Close Proximity Operations

4 Deimos-Phobos Transfer and Release of PM

5 Phobos Close Proximity Operations

6 Descent and Landing and Surface Operations

7 Ascent by ERV

8 Departure and Transfer Mars-Earth

9 Reentry with ERC

Figure 2-1: Baseline Mission Architecture

In Table 2-2, the mission phases and milestones with the corresponding durations and dates are listed for the short and long mission scenarios of the baseline launch date in 2024. The total mission duration is 2.7 years for the short mission scenario and 4.8 years for the long mission scenario, each including significant margin with respect to the available days before departure. A main advantage of the long mission scenario is to allow for a much longer Deimos characterisation phase, but this was not deemed necessary at this stage.

Milestones and Mission Phases	Date / Duration [d]	
Earth Departure Date	22/09/2024	12/10/2024
Launch and Direct Escape	0	0
Transfer Earth-Mars	352	332
Retained Mars Arrival Date (Worst case)	09/09/2025	09/09/2025
Transfer to Deimos	30	30
Deimos Close Proximity Phase	25	300
Deimos-Phobos Transfer	5	5
Phobos Close Proximity Phase	145	490
Descent and Landing Phase	14	14
Surface Operations Phase	6	6
Ascent Phase	2	2
Departure Phase	30	30
Sun-Earth conjunction (operations not possible)	50	100
Mars Departure Date to Earth	03/08/2026	06/09/2028
Transfer Mars-Earth	320	339
Re-entry Phase	0	0
Arrival to Earth	19/06/2027	11/08/2029

Table 2-2: Mission Phases and Milestones for Short (left) and Long (right) Mission Scenario and Launch in 2024

2.4 Phobos Sample Return Composite

The Phobos Sample Return composite is depicted in Figure 2-2. The main characteristics of the composite are listed in Table 2-3.

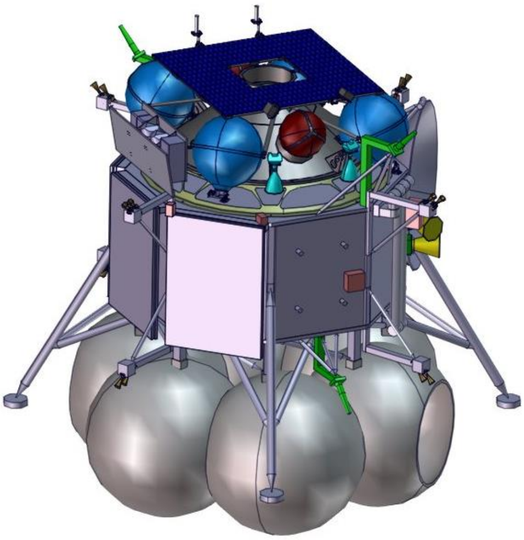


Figure 2-2 : Phobos Sample Return composite

Composite Main Characteristics	
Mass (incl. Margin)	Dry Mass: 1694 kg
	Science Instruments Mass: 38.4 kg
	Max Propellant Mass: 3377 kg (launch 2026)
S/C Main Components	- ERC (Earth Re-entry Capsule) - ERV (Earth Return Vehicle) - LM incl. science P/L (Lander Module) - PM (Propulsion Module)

Table 2-3: Composite main characteristics

2.5 Phobos Sample Return Elements

2.5.1 Earth Re-entry Capsule

The Earth re-entry capsule (ERC) and its main characteristics are presented in Table 2-4.

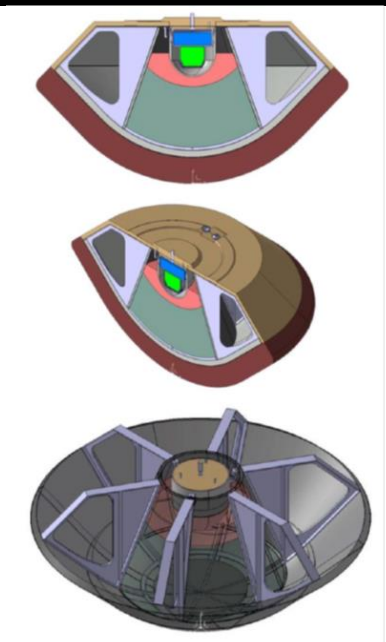
Earth Re-entry Capsule Description			
Trajectory	Landing location	Kazakhstan	
	Entry velocity	12.3 km/s (relative entry velocity - worst case retrograde)	
	FPA	-9.8 deg (nominal)	
	Mass	35 kg (incl. margin)	
Shape	Scaled from Hayabusa 45° half cone front shield		
	Main Diameter	0.75 m	
TPS	FS: ASTERM BS: Norcoat Liege		
	Heat load	Max: ¬ 221 MJ/m^2 (w. margin)	
	Heat Flux	Max: ¬ 15 MW/m^2 (w. margin)	
EDLS	None (no parachute)		
Structure	Load bearing Crushable materials to limit loads on sample		
Mechanisms	Sample container		
	Spin Separation device remaining on ERC		
GNC	None (uncontrolled re-entry)		
Communications	High g-load resistant recovery beacon based on aviation ELT or alternative		
DHS	None		

Table 2-4: Earth Re-entry Capsule main characteristics

2.5.2 Earth Return Vehicle

Table 2-5 shows the Earth return vehicle (ERV) and its major characteristics.

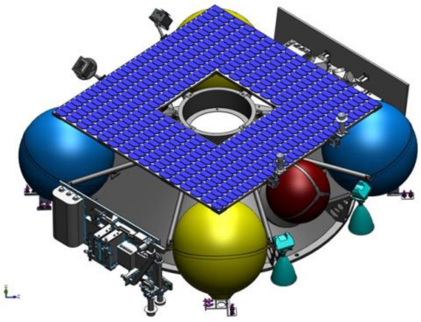
Earth Return Vehicle Characteristics			
AOCS/GNC	Sensors	Star Trackers	
		Sun Sensors	
		IMU	
	RCS	16 x 0.8N Thrusters, cold gas	
Propulsion	Bipropellant system, NTO/N2H4		
	Main Engine: 4 x 123.5N		
	Tanks: 4 fuel + 2 pressurant		
Power	SA	Body mounted	
	Battery	1 x Lithium Ion	
		BoL energy: 616 Wh	
	On-board voltage	27±1.35V	
Communications	All X-Band system		
	2 omni-directional antennas		
Thermal	MLI, heating lines, heaters		
DHS	OBC		
Mechanism	ERC spin separation device (SED TRP)		
	ERV separation remaining on ERV		
	Cable cutters		
	ERC ring hinge		
	ERC hold down		
Structure	Structural tanks + central cone for ERC accommodation		

Table 2-5: Earth Return Vehicle main characteristics

2.5.3 Landing Module

The Landing Module (LM) and its subsystem details are depicted in Table 2-6.

Lander Characteristics			
AOCS	D&L	Autonomous relative navigation	
	Sensors	2 x Star Tracker (AASTR)	
		2 x European IMU (Astrix 1090 + QA3000)	
		Wide Angle Cameras (2 OH) + (1EU), FoV: 53°	
		2 x Coarse Sun Sensor (TNO)	
		2 x Radar Altimeter	
	Actuator	4 x Reaction wheels (RSI 12/75-60)	
		16 / 24 x 20N thrusters	
Propulsion	Monopropellant system (Hydrazine)		
	Main engine: 1 x 1.1kN HTAE		
	Tanks	4 x Eurostar 2000 based, with 1801kg propellant	
Power	SA	5 x deployable wings	
		Solar cells: 30% 3J GaAs	

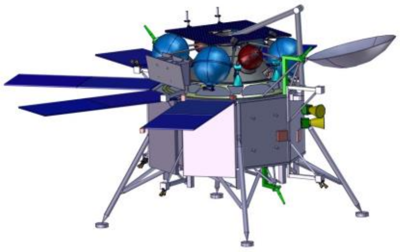
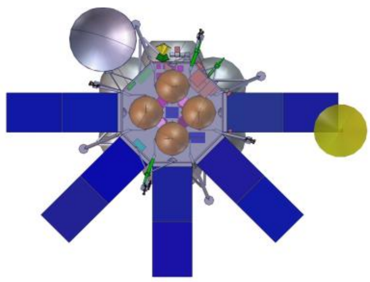
Lander Characteristics			
		Total area: 10.8 m²	 
		1.2 kW (EOL Mars Orbit)	
	Battery	1 x Lithium Ion	
		BoL energy: 2600 Wh	
	Bus	28V MPPT regulated bus	
Communications	All X-Band system		
	1 x steerable HGA		
	3 x fixed LGA for 4π coverage		
	2 x TWT Power: 65W		
	2 x optional LGAs on PM		
Thermal	MLI, heating lines, Black Paint, SSM, insulating Stand-Offs		
	No heat pipes		
Mechanism	Sample	1 x Robotic arm incl. gripper	
		4 x landing legs	
		Sampling and containment tool (Rotary brushes)	
	Support	SA HDRM	
		HGA pointing mechanism	
		HGA pointing electronics	
		HGA resettable HDRM (RUAG)	
		Robotic arm HDRM	
		ERV separation device	
		ERV ejection springs	
DHS	OBC + MM based on LEON-FT		
Structure	Octagonal structure with CFRP and Al-Al panels. Corner beams transferring the load from the 8 PM hard points; top and bottom covers		

Table 2-6: Landing Module main characteristics

2.5.4 Propulsion Module

In Table 2-7, the Propulsion Module (PM) is shown and its main characteristics are listed.

Propulsion Module Characteristics		
AOCS	None (Controlled by LM)	
Propulsion	Bipropellant system N2H2/NTO	
	Main engine: 20 kN	
	Tanks: 6 spherical	
Power	Chemical battery for propulsion power supply	
Communications	X-band antenna (Optional control by LM)	
Thermal	MLI, heaters	
DHS	None	
Structure	Structural tanks	

Table 2-7: Propulsion Module main characteristics

2.6 Conclusions and Options

The Phobos Sample Return study has demonstrated the feasiblity of a reference mission scenario for the baseline, with the following main characteristics.

- Russian PM and ERV design derived from the Phobos-Grunt mission, with a PM design almost recurrent from Fregat , and an ERV design tailored to meet the ESA ERC release requirements, as well as to be compatible with a potential future Mars Sample Return mission.
- European LM and ERC design derived from ESA Phootprint studies. The LM structural design has been optimised with respect to the interfaces with the Russian modules
- Outbound double stage transfer with staging at Phobos arrival after Deimos visit
- Joined European / Russian scientific payload in LM
- 2 options for sample acquisition have been investigated (one allowing bulk sampling, other allowing precise sampling)
- LM survives on surface after ERV departure
- Inbound single stage return transfer
- ERC re-entry and landing in Kazakhstan.

Furthermore, a backup scenario has also been studied (at a lower design definition level compared to the baseline) following the same mission architecture as the baseline but with a different cooperation scenario. Its main characteristics are as follows:

- ERC (ESA) and ERV (RU) are identical to the baseline mission
- Switch of responsibility between ESA and Russia on the PM and LM resulting in a different staging: Jettisoning of the PM at Deimos arrival (w.r.t. Phobos arrival in the baseline) and Deimos to Phobos transfer by the lander propulsion system
- Russian LM extensively based on Phobos-Grunt re-use
- ESA PM largely based on the MREP pre-Phase A Phootprint system studies design heritage (as well as Lisa Pathfinder mission).

Beyond demonstrating the technical feasibility of a combined Russian/European Phobos sample return mission, this study has proven that real-time concurrent activities can successfully be performed between ESA, IKI and Lavochkin. This study has been the first CDF study where ESA has collaborated in real time with the Russian partners IKI and Lavochkin and repeating this experience in the future can only be encouraged.

3 MISSION OBJECTIVES

3.1 Background

ESA-SRE is currently in discussion with the Russian Space Agency ROSCOSMOS for a potential common mission in the frame of Robotic Exploration of Mars and other bodies in the Solar System following cooperation agreement signed by both agencies in early 2013.

ExoMars 2016 and 2018 missions are considered as the first stage of this cooperation, but this agreement stipulates that subsequent stages of cooperation on Mars Exploration between the agencies could be undertaken.

This has been further developed during bilateral discussions which have concluded that Russian and ESA plans for Mars exploration framework were quite similar in several aspects. In particular both have Mars Sample Return (MSR) as long term objective, and both with an interest in a Phobos Sample Return mission as an intermediate mission to MSR.

A Joint ESA-ROSCOSMOS Working Group, with the mandate to build-up scenarios for Mars robotics cooperation beyond ExoMars, have concluded on various possible joint Phobos Sample Return mission scenarios and agreed on two possible options for responsibility sharing between Russia and Europe as well as on the need of jointly assessing the technical feasibility of these scenarios.

In March 2014, the heads of Agencies endorsed the joint CDF study to be conducted by IKI, Lavochkin, and ESA.

3.2 Science Objectives

The scientific objectives were under consolidation at the time of this study (ESA-IKI). The major aspects can be summarised as follow:

- To sample and return more than 100g of Phobos surface regolith
- To perform science characterisation of Phobos at global and local scale
  - Global characterisation by remote sensing of Phobos from Trailing & Quasi Satellite Orbits (QSO)
  - Fly-bys for high resolution measurements, local characterisation of candidate landing sites
  - In-situ imaging of the sampling area and sampling point at very high resolution before and after the sampling.
  - Post-sampling monitoring science done by the Landing Module which will remain on the Phobos surface after the Earth Return Vehicle has lifted-off
- To perform science characterisation of Deimos at global scale only
  - Remote sensing of Deimos from Trailing & Quasi Satellite Orbits.

3.3 Technology Objectives

In a technology standpoint, the following objectives have been set:

- To demonstrate or mature technologies required for Mars Sample Return:
  - Earth Re-entry Capsule, Earth Return Vehicle, Autonomous Rendezvous in Mars orbit (in some ways similar to a landing on Phobos), Sampling, Sample Handling and Sealing, Operations.

3.4 Options

As introduced in section 3.1, among the various cooperation scenarios identified by the joined ESA-ROSCOSMOS Working Group, the following two options have been selected as main cooperation scenarios.

	Baseline		Backup	
	ESA	ROSCOSMOS	ESA	ROSCOSMOS
Launcher				
Transfer Propulsion Module (PM)				
Landing Spacecraft (LM)				
Sampling and Transfer Equipment		With ROSCOSMOS participation		With ROSCOSMOS participation
Earth Return Vehicle (ERV)				
Earth Re-Entry Capsule (ERC)				
Science Instruments				
Launch Ops				
SC Cruise + Landing Ops				
ERV SC Operations				
Science Operations				
Ground support				
Sample Receiving facility				
Science exploitation				

Table 3-1: ESA-ROSCOSMOS Cooperation scenario

This follows the following model space segment elements:

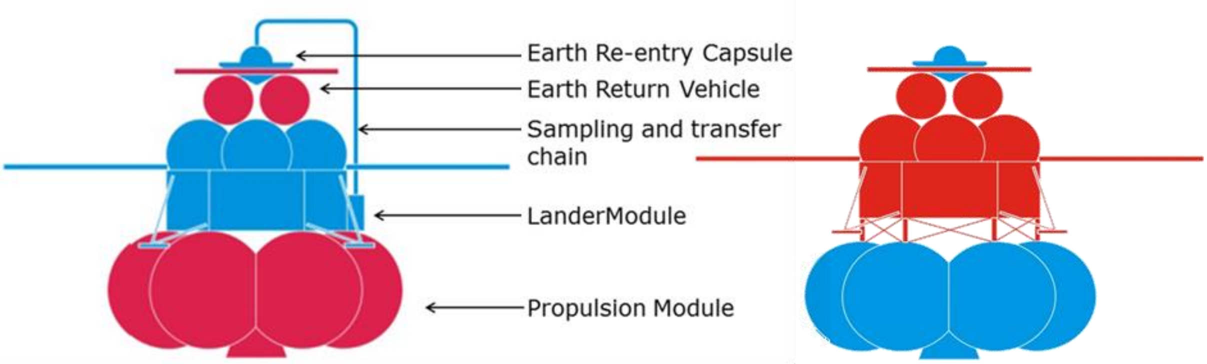


Figure 3-1: Elements cooperation scenario (left: baseline, right: backup)

3.5 Study Objectives

This CDF study has focused on the feasibility evaluation of the baseline cooperation scenario previously presented.

As a secondary task, the backup cooperation scenario has been addressed at system level.

For each cooperation scenario , both Russian and European sampling chain solutions have been considered (sampling tool and robotic arm).

3.6 Mission and System Requirements

The Mission and System Requirements for this study have been compiled in the "Phobos Sample Return ROSCOSMOS-ESA joint mission, Mission Requirement Document" RD[1] (referred to in this report as the "MRD").

This document has been prepared by the study's customer starting from the ESA Phootprint MRD and further elaborated and agreed with all study participants (IKI, Lavochkin, ESA-SRE & ESA CDF).

All refined requirements are presented in chapter 8 – Systems.

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4 MISSION ANALYSIS

4.1 Requirements and Design Drivers

The prime task of Mission Analysis in the context of the present study was to establish the changes in the mission constraints and the mass and delta-v budgets if the launch vehicle is no longer a European Ariane 5 ECA, as assumed in RD[2] for the MREP pre-Phase A Phootprint system study, but a Proton M / Breeze M provided by Russia and launched from Baikonur, which is the baseline solution agreed for this study.

The launch opportunities for short (around one half ellipse) and long (around one and a half ellipses) transfers and also short transfers preceded by a one year Earth-Earth arc and an Earth swing-by in the 2022-2028 time frame are to be taken into account (to provide a wider view than the baseline 2024-2026 slot).

4.2 Assumptions and Trade-Offs

The transfer delta-v costs for a targeted first encounter with Phobos were to be traded against the case where Deimos is targeted first. In that case, also the delta-v cost and duration of a transfer from Deimos to Phobos is to be assessed.

4.3 Baseline Design

4.3.1 Launch and Earth Escape Sequence

Unlike launch with Ariane 5, Soyuz or Atlas V, the Proton-M/Breeze-M escape sequence can be very lengthy, requiring four (sometimes five) burns of the 20 kN motor on the Breeze-M stage.

The first of the Breeze M manoeuvres inserts into a 185 km parking orbit, while burns 2 and 3 inject into consecutively more highly eccentric orbits. Between burns 3 and 4, Breeze M jettisons an empty toroidal tank. Finally, burn 4 injects into hyperbolic escape.

This means that at least two complete revolutions on eccentric orbits plus the escape arc are performed before escape, leading to five passes through the Van Allen belts and incurring radiation doses. The length of the sequence also has impacts on the power and thermal subsystems, and there can be multiple eclipse passes. The following table is extracted from the Proton M user's manual RD[3] Annex F.

Hyperbolic Excess Velocity, V∞ (m/s)	PSM (kg)	Injection Time (hrs)
0	6475	8.1
500	6445	8.1
1000	6355	8.2
1500	6205	8.4
2000	6002	8.7
2500	5748	9.0
3000	5454	9.5
3500	5124	10.2
4000	4745	11.0
4500	4361	12.2
5000	3971	13.8
5500	3556	15.0
6000	3111	15.0
Performances have been calculated for the standard PLF (15,255 mm long). FMHF at the PLF jettison does not exceed 1135 W/m^2. PSM includes LV AS mass. PSM has been defined for the US 2.33σ propellant margin.		

Table 4-1: Proton M / Breeze M Escape Performance and Duration

4.3.2 Outbound Transfers

The following timeline is followed by a spacecraft launched by Proton M / Breeze M until it reaches the orbit of Phobos (or Deimos):

- Launch + escape sequence
- Launcher injection correction up to 7 days after launch
- None, one or several deterministic (large) manoeuvres during transfer (Deep Space Manoeuvre = DSM 1, 2, 3...)
- (Earth swing-by for some regarded cases)
- Mars orbit insertion: Injection into inclined HEO around Mars with period of 4 sols, apoares altitude ~96,000 km, periares altitude ~300 km.
- Target orbit acquisition (TOA) 1 near apoares: raises periares to Phobos (or Deimos) orbit altitude, reduces inclination
- Target orbit acquisition (TOA) 2 at new periares (=Phobos (or Deimos) orbit altitude: circularizes orbit and adjusts inclination
- Spacecraft is now in an orbit that matches that of Phobos (or Deimos), but some phasing (catching up) may still be required.

4.4 Budgets

It is assumed that an individual launcher program will be provided for each day of the 21 day launch period (LP). In practice, this might not be so. If the number of launcher programs is limited to e.g., 3 per launch period, the entire scenario would have to be re-assessed. Typically, this will lead to penalties – increases in delta-v, loss in wet mass or both. In the following tables, the lowest launcher performance throughout the LP was applied to the entire LP. The mass budget is estimated subject to this value.

4.4.1 Short Transfers

Case	P22S		P24S		P26S		P28S	
LPO/LPC	LPO	LPC	LPO	LPC	LPO	LPC	LPO	LPC
Launch date	2022/08/27	2022/09/16	2024/09/23	2024/10/12	2026/10/21	2026/11/10	2028/11/12	2028/12/02
v-infinity [km/s]	3.637	3.747	3.489	3.452	3.138	3.229	3.21	3
declination [deg]	7.6	11.3	13.8	17.6	23.5	37	20.1	35.1
launcher performance [kg]	5019	4936	5130	5155	5362	5302	5315	5454
wet mass [kg]	**4826**		**5020**		**5192**		**5205**	
DSM date	2023/01/13	2023/02/27	-	-	-	-	-	-
DSM [m/s]	297	203	0	0	0	0	0	0
arrival	2023/08/04	2023/09/07	2025/08/22	2025/09/09	2027/09/04	2027/09/09	2029/09/14	2029/09/26
v-arrival [km/s]	2.239	2.602	2.421	2.525	2.593	2.568	2.978	2.978
First Target: Phobos								
MOI impulsive [m/s]	591	746	661	709	741	730	934	935
TOA-1 [m/s]	213	160	105	103	120	146	152	191
TOA-2 [m/s]	785	755	753	753	755	755	755	756
Total delta-v w/o GL [m/s]	1886	1864	1519	1565	1616	1631	1841	1882
Duration	350	356	333	332	318	303	306	298
Dry mass w/o GL [kg]	**2636**	**2655**	**3085**	**3040**	**3093**	**3078**	**2885**	**2847**
First Target: Deimos								
MOI impulsive [m/s]	804	962	779	823	879	917	1156	1216
TOA-1 [m/s]	279	284	257	262	270	279	294	299
TOA-2 [m/s]	403	401	379	380	383	391	408	478
Total delta-v w/o GL [m/s]	1783	1850	1415	1465	1532	1587	1858	1993
Duration	350	356	333	332	318	303	306	298
Dry mass w/o GL [kg]	**2725**	**2667**	**3189**	**3139**	**3177**	**3122**	**2869**	**2748**

Table 4-2: Summary of Short Transfers, LPO and LPC

4.4.2 Long Transfers

Case	P22L		P24L		P26L		P28L	
LPO/LPC	LPO	LPC	LPO	LPC	LPO	LPC	LPO	LPC
Launch date	2021/11/11	2021/12/01	2024/05/31	2024/06/20	2026/06/14	2026/07/04	2028/05/30	2028/06/19
v-infinity [km/s]	2.908	3	2.886	3.131	3.02	3.307	2.478	2.556
declination [deg]	25	35.1	-11.6	1.822	-1	10.3	-11.8	-0.1
launcher performance [kg]	5507	5454	5520	5367	5440	5251	5759	5715
wet mass [kg]	**5344**		**5257**		**5140**		**5203**	
DSM date	2023/06/12	2023/09/09	2024/12/23	2025/01/25	2027/01/13	2027/03/26	2029/06/29	2029/09/22
DSM [m/s]	172	40	1085	973	237	148	425	490
arrival	2024/01/20	2024/01/31	2026/07/15	2026/08/01	2028/06/17	2028/07/13	2030/04/18	2030/05/06
v-arrival [km/s]	2.872	3.007	1.676	1.707	2.442	2.459	2.738	2.649
First Target: Phobos								
MOI impulsive [m/s]	879	939	370	380	671	678	811	767
TOA-1 [m/s]	150	190	98	98	99	97	103	101
TOA-2 [m/s]	755	756	753	753	753	753	753	753
Total delta-v w/o GL [m/s]	1956	1925	2306	2204	1760	1676	2092	2111
Duration	350	791	775	772	734	740	688	686
Dry mass w/o GL [kg]	**2855**	**2883**	**2510**	**2594**	**2924**	**3004**	**2661**	**2645**
First Target: Deimos								
MOI impulsive [m/s]	1097	1233	489	499	781	786	928	880
TOA-1 [m/s]	275	292	193	196	256	257	273	268
TOA-2 [m/s]	449	471	372	372	379	379	382	381
Total delta-v w/o GL [m/s]	1993	2036	2139	2040	1653	1570	2008	2019
Duration	800	791	775	772	734	740	688	686
Dry mass w/o GL [kg]	**2821**	**2782**	**2648**	**2734**	**3026**	**3107**	**2733**	**2724**

Table 4-3: Summary of Long Transfers, LPO and LPC

4.4.3 Earth Swing-by Transfers

ESB transfers have been assessed. They do not lead to an increase in mission performance for the regarded scenarios.

4.4.4 Near Phobos Operations

Near-Phobos operations have not been revised in the context of the present study. The information given in RD[2] remains applicable.

4.4.5 Earth Return

All operations required to leave Phobos and return to the Earth have not been revisited in the context of the present study. The information given in RD[2] remains applicable.

4.5 Options

4.5.1 Transfer from Deimos to Phobos Orbit

If the spacecraft were sent to the Deimos orbit at first, a subsequent transfer to the Phobos orbit would be required. This would add a delta-v penalty in the order of 800 m/s to the delta-v budget of the outbound mission. The duration of the transfer is around 9 hours, however lengthy drift periods before and after the transfer can be required.

Alternatively, it would be possible – with much less delta-v penalty– to perform a series of close fly-bys at Deimos by splitting TOA2 into several parts and first inserting into an intermediate orbit that is synchronous with that of Deimos. On that orbit, a number of fly-bys could be targeted before the apoares is reduced to the Phobos orbit radius.

5 PAYLOAD

5.1 Requirements and Design Drivers

The selection of candidate instruments of the model payload is largely defined by science requirements given in RD[4], the Science Requirements Document (SciRD). This document originates from a previous mission study called Phootprint. This mission addresses in a very similar manner the return of sample material from Phobos to Earth. Therefore the SciRD has been adopted by all participating parties to this study with the addition of Deimos global characterisation (similar requirements as per Phobos global characterisation).

However, the payload model differs from the one selected for Phootprint and for this study, the payload definition was further discussed and agreed between ESA and Roscosmos / IKI (in particular few instruments were added compared to Phootprint in order to increase the science achievements). Consequently the suite of payload instruments has been modified to the current mission cooperation scenario and design and is described in the following sections.

5.2 Assumptions and Trade-Offs

A variety of payload candidate instruments have been suggested for the Lander Module (LM), for the Earth Return Vehicle (ERV) and the Earth Re-entry Capsule (ERC). In order to establish a model payload, candidate instruments have been prioritised in three different categories mounted on the LM, ERV or ERC.

1. Category I (instruments mounted on LM only)
    a. Payload instruments of essential importance to fulfil the agreed scientific objectives
    b. These instruments shall be accommodated in the mission and spacecraft design and are part of the model payload.
2. Category II (instruments mounted on LM only)
    a. Payload instruments that provide additional scientific data and would improve the science return of the mission. However, these results are not mission critical to the mission success.
    b. They are not part of the model payload, they are optional in case a resource budget becomes available for proper integration.
3. Category III (instruments mounted on ERV or ERC)
    a. Payload instruments that provide additional scientific information however do not fit in category 1 or 2.
    b. Accommodation in mission and spacecraft design is subject to study.

5.3 Baseline Design and Characteristic of Model Payload Instruments; Category I (Instruments Mounted on LM Only)

This section describes the key design parameters and their scientific relevance for each instrument of the model payload.

The instruments take strong heritage from previous space missions, instrument proposals to missions in implementation or under study i.e. Phobos-Grunt, ExoMars, MarcoPolo and MarcoPolo-R.

Generally the instrument details have a confidential character and cannot be displayed in this report. Therefore the instrument parameters are generalised but still leading to a robust estimate of their resource budgets for the purpose of this study.

5.3.1 Wide Angle Camera - WAC

The WAC is based on a dioptric design with a focal length of 105 mm. The detector with 2048x2048 pixels (A/D 16 bit) covers the wavelength range between 400 and 950 nm. The resource budgets (Table 5-1) are based on a camera design proposed for the MarcoPolo assessment study.

The WAC is used for overview images to locate the images obtained by the NAC. Further limb observation will support the re-construction of the shape model of Phobos.

5.3.2 Narrow Angle Camera – NAC and CSU

The NAC is based on a three mirror anastigmatic optical design with a focal length of 660 mm. The 2048x2048 pixel detector (16 bit A/D) covers the visible and near infrared wavelength range (400-850 nm). The operational distance of the NAC between ~5km distance and infinity to the surface requires the introduction of a focus mechanism to the design. Further it is assumed that the NAC is operated as a framing camera.

The generic resource budgets (Table 5-1) are obtained from a camera design proposed for the MarcoPolo-R mission study.

The NAC is responsible for the global characterisation and mapping of Phobos as well as the landing site characterisation (i.e. Phobos local characterisation).

The WAC and NAC will share a Common Support Unit (CSU) responsible for power conditioning, command and data acquisition and handling.

5.3.3 Stereo Camera – StereoCam

The StereoCam consist of two camera heads mounted in a single housing including the electronics. The unit is placed outside on the lander platform viewing the landing site and the working area of the robotic arm. The CCD consists of a 1024x1024 pixel matrix with fast read out of up to 30 frames per second.

The StereoCam is used for the sampling area characterisation, identification of the sampling point(s) and observes the robotic operations.

The instrument has been developed for the Phobos-Grunt mission (Table 5-2).

5.3.4 Close-Up Imager – CLUPI

The CLUPI is a lens camera in a highly integrated design. A colour sensor takes full RGB images without any loss in resolution. Through a focus mechanism the camera can be operated from infinity down to a few cm distance to the target. It is located on the robotic arm in close proximity to the sampling tool.

The characterisation of the sampling point(s) at sub-mm resolution before and after the sample is collected is the main scientific goal of this instrument.

The generic resource budgets (Table 5-1) are obtained from a camera design proposed for the MarcoPolo-R mission study.

5.3.5 Visible Near Infrared Spectrometer – VisNIR

The VisNIR is an imaging spectrometer. It utilizes a slit spectrometer based on a prism system dispersion in an Offner relay configuration. The spectrometer covers the spectral range from 0.4 to 4.0 µm. A pointing mirror mechanism at the front of the instrument provides some flexibility during instrument operations at the target. The detector is thermally controlled by a cryo cooler.

The generic resource budgets (Table 5-2) are obtained from a camera design proposed for the MarcoPolo-R mission study.

The VisNIR provides global and local mapping and landing site characterisation by mineralogical composition and distribution of provinces on the surface of Phobos.

5.3.6 Mid Infrared Spectrometer – MidIR

The MidIR is a spectrometer and thermal imager. It contains two light paths separated by a flip mechanism which either exposures the imager or spectroscopic unit. A double entrance with corresponding pointing mirror allows in-flight calibration by looking into deep space. An uncooled bolometer simplifies the design and makes an active cooler obsolete. The instrument covers the wavelength range between 8 µm and 18 µm.

The generic resource budgets (Table 5-2) are obtained from a camera design proposed for the MarcoPolo-R mission study.

The instrument delivers a global temperature map and identifies mineral features and regolith physical properties on a global scale as well as of the landing site characterisation.

5.3.7 Neutron Spectrometer - ADRON-RM

The instrument measures passively thermal and epithermal neutrons released by the surface of Phobos and also from the general radiation environment. The design is based on ^3He proportional counters.

A similar instrument is under development for the ExoMars rover mission. The generic resource budgets are given in (Table 5-2).

ADRON-RM provides data on the hydrogen content in the upper layer (~1 m) of Phobos. The hydrogen is bound in water, water ice and as $(OH)^-$ and H_2O molecules in hydrated minerals.

5.3.8 Dust Detector - DIAMOND

The dust counter is based on a charged grid and an underlying piezoceramic plate detecting the impact of the particle. Through a direct measurement the impulse, velocity and direction of the incoming particle can be determined.

Two identical and independent instruments are mounted fore- and backward looking with respect to the flight direction.

An identical instrument was build for the Phobos-Grunt mission. The generic resource budgets are given in (Table 5-2).

5.3.9 Starfield Observation – LIBRATION

The LIBRATION experiment is based on a wide field, startracker like camera. It is operated during and after landing operations. As such it is mounted looking upwards observing the star field. It should remain on the surface module for continued operations after lift-off of the Earth return vehicle.

This instrument was build for the Phobos-Grunt mission. The generic resource budgets are given in (Table 5-2).

The continuous observation of stars from the surface of Phobos reveals a subtle wobbling of Phobos. This refers to irregularities on the inner structure.

5.3.10 Radio Science Experiment

The radio science experiment is linked to the spacecraft communication subsystem. Currently no specific requirements beyond its standard performance are identified.

5.4 Optional Instruments; Category II (Instruments Mounted on LM Only)

The optional instruments consist of a less precise specified list of possible instrumentation. It mainly refers to instruments that perform in-situ observation or require the acquisition of a surface sample. Often auxiliary like a robotic arm or sample processing machinery is required for a successful installation of the instruments.

Typical examples of this group measure the chemical and isotopic composition by means of:

- X-ray fluorescence
- Laser-plasma spectroscopy
- Various mass spectrometry techniques.

The mineralogical composition and specific phase identification can be tackled by:

- Moessbauer spectrometry
- Raman spectrometry
- X-ray diffraction spectrometry
- Micro-infrared spectroscopy
- Attenuated total reflection spectrometry.

A variety of penetrator type devices can assess physical properties of the upper regolith layer. In addition, these devices can be equipped with analytical tools of the examples above.

5.5 Optional Instruments Category III (Instruments Mounted on ERV and/or ERC)

5.5.1 BIOPhobos Experiment

This unit was designed to investigate the dormant forms of biological objects during long term interplanetary missions to deal with planetary quarantine and astrobiology issues. The results of this experiment can be useful in biomedicine and manned space exploration. A small hermetic capsule (about 100 g) is mounted into the returned ERC without any actions during the flight to Mars and back. For further details see Table 5-3.

5.5.2 TV Camera

A TV camera is mounted on the ERV to take pictures (video) of the LM at the surface of Phobos during take-off. These pictures would be very useful for references of measurements and sampling on the surface and for public relation purposes. For further details see Table 5-3.

5.6 List of Equipment

In Table 5-1 and Table 5-2 the modal payload is listed. The key instruments characteristic resource requirements are summarised.

	WAC	NAC	CSU	Stereo Cam	CLUPI
S/C interface					
Accommodation Operation	Inside Orbit	Inside Orbit	Inside Orbit	Lander Surface	Arm Surface
electrical	na		28 V reg.	28 V reg	na
data	na	na	spacewire	RS-232	na
thermal	radiator	radiator			radiator
Pointing					
direction	*nadir*	*nadir*	*Close to WAC/NAC*	30° to normal	*According arm pos.*
absolute error [mrad]	1.25	1.25	na	n/a	8.7
Relative error (stability) [µrad/s]	200	15	na	n/a	20
Field of view	11.2	1.7	na	70° of each camera and 60% stereo overlapping	14.0
Unobstructed FoV	±45	±45	na	n/a	Tbd
Physical					
No. of unit	1	1	1	2 cameras in one case	1

	WAC	NAC	CSU	Stereo Cam	CLUPI
Volume (hxwxl) [mm]	115x172x237	380x340x200 body 180x130Ø baffle	100x220x290	230x150x180	120x100x225
Mass [kg]	2.15	7.0	3.5	1.5	0.7
Mass +20%	2.65	8.4	4.2	1.8	0.84
Power [W]					
Orbit operations	11.5	15.6	8	n/a	na
Surface operations	na	na	na	9	12.5
Stand-by	8 tbc	8 tbc	8	1.5 TBC	3.3
Temperature [C°]					
Min/max ops	-55/+50	-55/+50	-55/+50	-40/+40	-20/+50
Min/max non ops	-55/+60	-55/+60	-55/+60	-60/+60	-30/+60
TRL	4	4	4		4

Table 5-1: Model Payload: camera systems

	VisNIR	midIR	ADRON-RM	DIAMOND	LIBRATION	RSE
S/C interface						
Accommodation operation	Inside Orbit	Inside Orbit	Bottom side Orbit/surface	Lander side Orbit/surface	Top deck surface	TT&C orbit
electrical	28 V reg.	28 V reg.	25-35 V reg	27 V reg	28 V reg	
data	spacewire	spacewire	TBD	RS-485/space ware possible	RS-422	
thermal	radiator	radiator	Under MLI	Under MLI	n/a	
Pointing						
direction	nadir	nadir	Lower hemisphere	Two opposite sides of the LM	Upper hemisphere	
absolute error [mrad]	0.12	1.6	n/a	n/a	n/a	
Relative error (stability) [µrad/s]	97	242	n/a	n/a	n/a	
Field of view	1.83	9.5x7	hemisphere	hemisphere	±18	
Unobstructed FoV	±15	11x11	n/a	n/a	±60	
Physical						
No. of unit	2	1	1	2	1	
Volume (hxwxl) [mm]	120x175x 357 opt.	14x25.5x2 9	210x205x60	150x150x80	65x60x100	

	VisNIR	midIR	ADRON-RM	DIAMOND	LIBRATION	RSE
	90x180x1 90 elec.					
Mass [kg]	7.10	6.3	1.6	1.9	0.25	
Mass +20%	8.52	7.6	1.9	2.3	0.3	
Power [W]						
Orbit operations	20	18	5	10	n/a	
Surface operations	na	na	5	10	2	
Stand-by	7	4	1	0.1 TBC	0.1 TBC	
Temperature [C°]						
Min/max ops	-20/+50	+5/+15	-20/+40	-20/+40	-40/+50	
Min/max non ops	-30/+60	-40/+40	-30/+50	-50/+60	-50/+60	

Table 5-2: Model Payload: Spectrometers and other instruments

In summary, the spacecraft should provide accommodation for the suite of model payload (Table 5-1 and Table 5-2) with the following total budget of:

- 32 kg applying a maturity margin of 20%, a total mass of 38.4 kg
- 80 W, applying 20% of maturity margin, a total of 96 W for orbit operations
- 38.5 W, applying 20% of maturity margin, a total of 47 W for surface operations.

	TV Camera	BioPhobos
S/C interface		
Accommodation Operation	ERV	ERC
electrical	28 V	na
data		na
thermal	radiator	na
Pointing		
direction	na	na
absolute error [mrad]	na	na
Relative error (stability) [µrad/s]	na	na
Field of view	60°x60°	na
Unobstructed field of view	60°x60°	na
Physical		
No. of unit	1	1
Volume (hxwxl) [mm]	80x80x80	
Mass [kg]	0,5	0.1
Mass +20%	0,6	0.11
Power [W]		
Orbit operations		na
Surface operations	na	na
Stand-by	na	na

	TV Camera	BioPhobos
Temperature [C°]		
Min/max ops	-20/+50	
Min/max non ops	-30/+70	
TRL		

Table 5-3: Optional Payload mounted on ERV and ERC

The data volume generated by the instruments during the different mission phases are specified in section 18.2.3.

Regarding the Science requirements for the Deimos and Phobos global and local characterisation, the resolution performance of the remote sensing instruments has to be analysed in further phases and discussed with the scientist considering the current mission operations and design constraints.

5.7 Technology Requirements

Generally the chosen candidates of the model payload suite have a strong heritage from previous missions or are at an advanced level of development. Certainly all instruments would require an adaptation to the specifics of a Phobos sample return mission. Yet no items on sub-system or component level could be identified that would be subject to a completely new development campaign.

Possibly critical items are rather subject of standard engineering work or candidates of a qualification campaign. Among those groups the following elements can be identified:

- High performance sensors
- Focus mechanisms
- Pointing mechanisms
- Automated covers.

6 PROPULSION MODULE (BASELINE)

6.1 Study Assumptions

During the joint Phobos-SR study the PM was considered as an executive module which is controlled by the LM. The purposes of this module are:

- To brake the composite SC in Mars gravisphere for acquisition of Mars orbit
- To perform high dV manoeuvres between Phobos and Deimos observation orbits.

Reasons to use PM as a controlled executive module are:

1. LAV has extensive experience in the development and operation of the PM itself as well as its prototype.The principles and materials used in the development of the PM have a long flight heritage and are applied in the newly developed Russian spacecraft
2. PM has wide opportunities for modification in terms of change of max tanks volume and structure strength increasing
3. PM has a simple and light-weight structure, allowing it to carry a lot of propellant with a minimum of incidental mass.

During bilateral discussions, several design options of the PM were reviewed for the Phobos Sample Return mission. The main factor which defined the final result was a required volume of propellant to achieve the purposes described above. Both parties concluded that it looks more reasonable to separate PM from LM at as late mission phase as possible, namely before landing on Phobos. This fact is due to decreased LM mass and also to reduce requirements for the LM propulsion capabilities.

6.2 Design Description

PM for the Phobos Sample Return mission has a structure of 6 intersecting spheres, 4 of them being bipropellant tanks (NTO/N2H4). Max propellant volume of the chosen PM option is 5300 kg, dry mass of the PM is about 650 kg. General view of the PM is shown in Figure 6-1.

Figure 6-1: PM general view

Propulsion module includes the following systems:

- Propulsion system, control unit
- TCS , control unit
- Chemical current source.

A key feature of the PM should be considered the absence of a control system, ACS and solar panels. The PM is equipped with a central engine with two thrust modes F1=19.85 kN and F2=14.00 kN, Isp=332s RD[37]. Moreover, it is worth noting that low-gain antenna on –X direction should be placed below the plane of the junction between PM and Launcher adapter to ensure the Earth link with a composite SC during mission phases before PM separation. This fact is due to the dimensions of the PM, which could be the cause of radio interference.

The PM separates from LM with an adapter. 8 spring-loaded actuators with pyrotechnical triggers are used for PM separation from LM with a force of 882N. Signal to pyrotechnical triggers for the synchronous operation comes from LM. It is necessary to use special pyrotechnical knife for the mechanical separation of the waveguide to –X antenna in case of PM separation. The pyrotechnical knife cutting process scheme is shown in Figure 6-2.

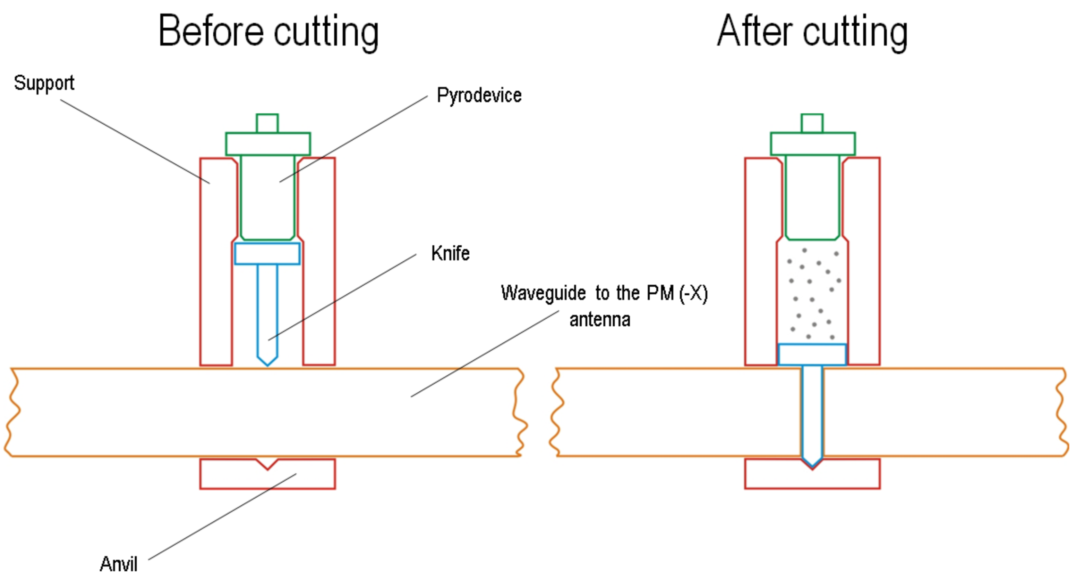


Figure 6-2: Pyrotechnical knife cutting process scheme

Proposal for PM power supply scheme and PM power consumption are shown in Figure 6-3 and Table 6-1 respectively.

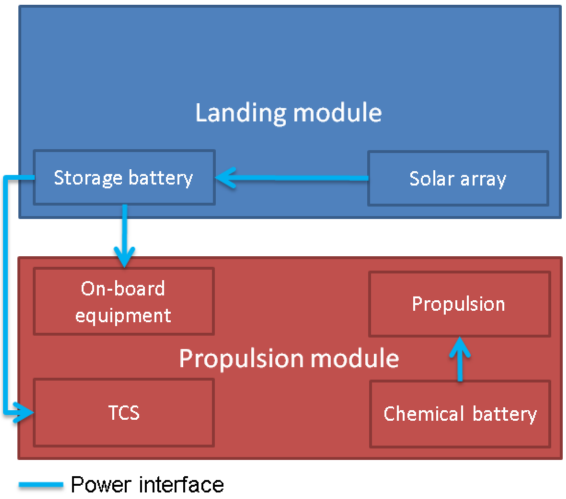


Figure 6-3: Proposal for PM power supply

		Propulsion module		
		On-board equipment	TCS	Propulsion
Propulsion	OFF	~20W	~280W	0
	Injection	~40W	~280W	~930W

Table 6-1: PM Power consumption

6.3 Options

PM options which have been discussed during joint CDF study are shown in Table 6-2.

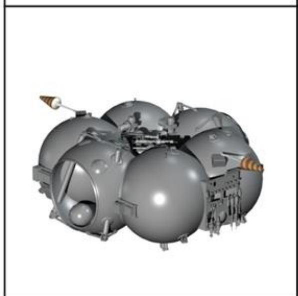
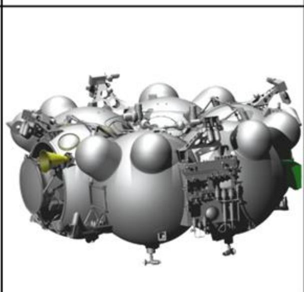
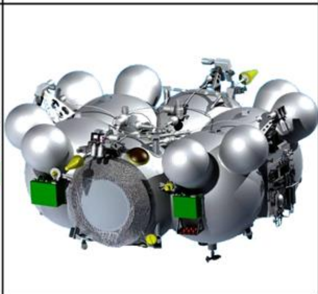
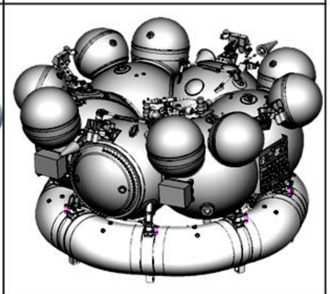
Option 1	Option 2	Option 3	Option 4
			
Baseline	Baseline with small extra tanks	Baseline with large extra tanks	Baseline with large extra tanks and Jettisonable ring-type tank
Max fuel 5 300 kg	Max fuel 5 800 kg	Max fuel 7 100 kg	Max fuel 10 250 kg

Table 6-2: PM Options

The analysis showed that in order to meet the requirements described in MRD, it is reasonable to use the first PM option. This option provides the possibility to fill the PM with the necessary amount of propellant to cover all high dV maneuvers during all mission phases before landing on Phobos.

6.4 Interface Requirements

During the joint CDF study, parties discussed mechanical and electrical interface requirements.

Regarding the mechanical interfaces, the parties have agreed on preliminary mounting dimensions and also on responsibility sharing scenario for the adapters and the separation systems.

The adapter and the separation system between LM and PM are under ROSCOSMOS responsibility, adapter mass is 35 kg, general view of the adapter and its mounting dimensions are shown in Figure 6-4.

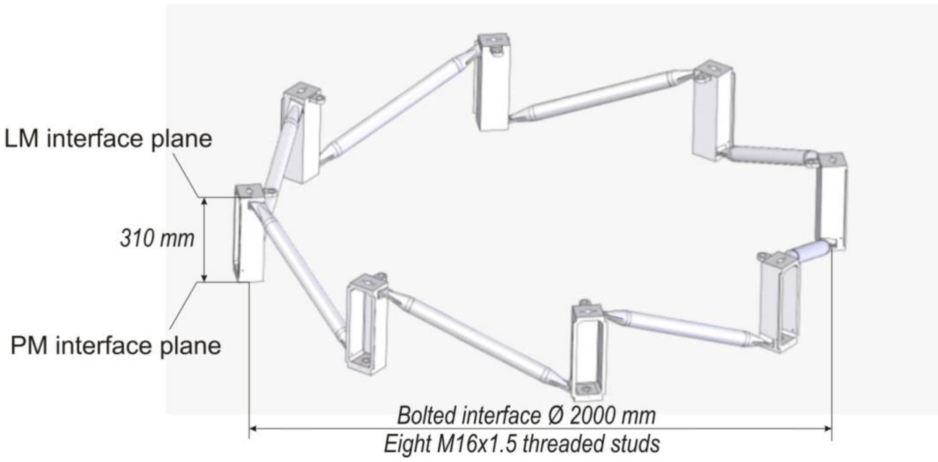


Figure 6-4: LM/PM adapter

Considering the options for data , electrical and RF interfaces - 2 options were presented. The first option is to control the PM from the LM. In this case we have all the 3 types of interfaces between LM and PM. Implementing that option means that ESA should provide power supply and PM control, and should also be responsible for separation command and for the commands to cut cables and waveguide to –X low-gain antenna.

The second option is to control the PM from the ERV, while the power for PM on-board equipment and TCS should be provided by LM EPS. Increasing power capabilities of ERV EPS for PM power supply would lead to a significant ERV mass increase, which is unacceptable in return mission phase. This approach therefore results into two types of interfaces between ERV and PM: Data and RF. These interfaces will provide control of the PM and communication with the composite SC. Graphic illustration of the schemes described above is shown in Figure 6-5.

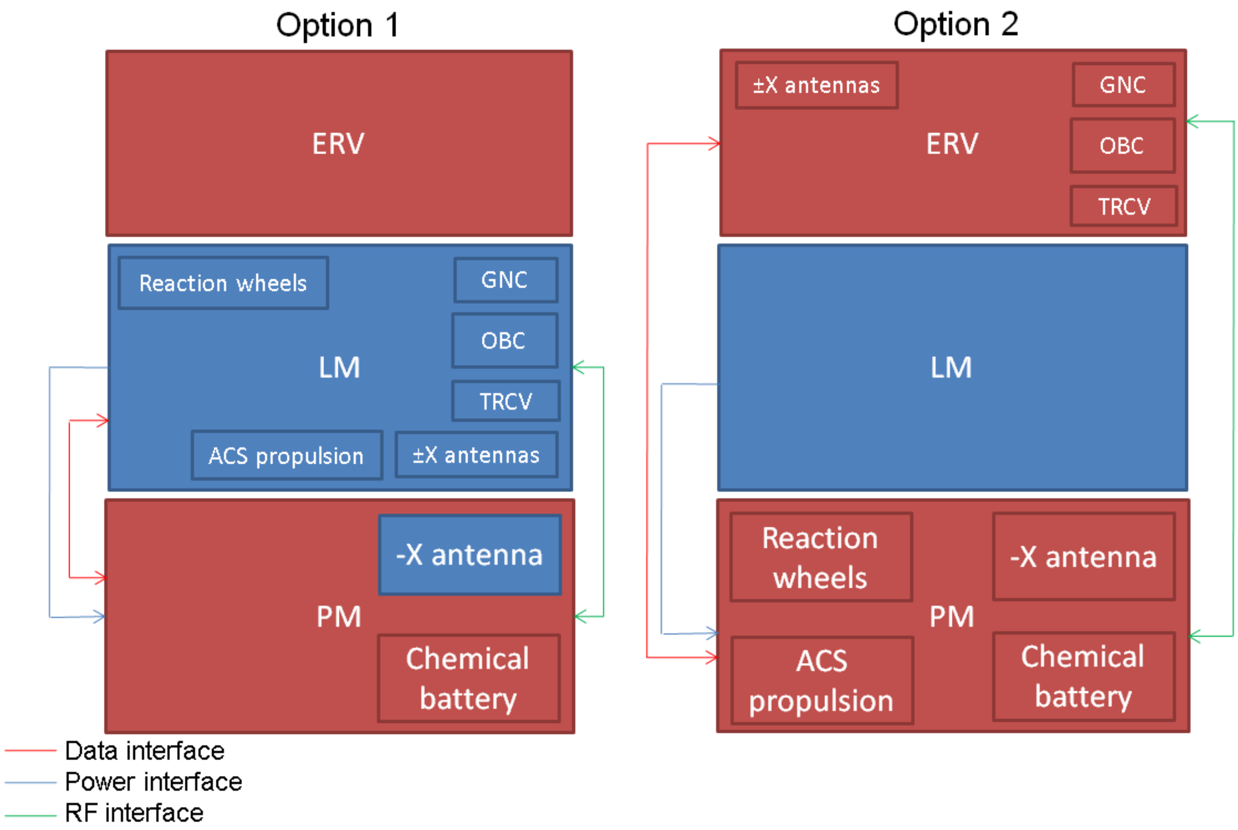


Figure 6-5: PM control options

The second PM control option is preferable from the control logic point of view. On the other hand, this option is less preferred because of mass increase and complexity of the mechanical separation systems. The great length of cables and waveguides, which are connecting ERV and PM, is the reason for mass increase. Another disadvantage appears because of the necessity of mechanical cutting cables and waveguide in not one, but in two points. This fact generates an additional problem of spacecraft removal from a collision with a cut portion of cables and waveguides.

6.5 Technology Requirements

Technical readiness level of PM elements are shown in Table 6-3.

Element		TRL status
1	**Propulsion system**	9
1.1	Main propulsion engine	9
1.3	Propellant storage and transfer system	9
2	**Electrical powersubsystem**	8

Element		TRL status
2.1	Chemical battery	9
2.2	Energy conversion equipment	8
3	**TCS**	8
4	**Separation system**	9
5	**Structure**	9

Table 6-3: PM Elements TRL

7 EARTH RETURN VEHICLE

7.1 Study Assumptions

During the joint Phobos-SR CDF study the ERV was considered as a newly developed product for the mission, largely based on the Phobos-Grunt heritage and on the subsequent Boomerang study. The purpose of the ERV is to deliver Phobos samples (which are located inside the ERC) to the Earth. It was agreed that the ERC mass shall be about 35 kg including margins. During bilateral discussions, several design options of the ERV were reviewed for the Phobos Sample Return mission. The main drivers that have influenced the final result were:

- ERC design (ESA responsibility)
- Number of ERC onboard
- Design of the system to put the samples inside the ERC
- ERV design unification for MSR mission (possibility to deliver ~120 kg ERC from Mars orbit to the Earth).

When considering options for ERC placement on-board ERV, the Russian side, using their previous experience, proposed several options which envisaged a separation system located in the area of the ERC TPS stagnation point. Moreover, the study considered not only options with one ERC onboard, but with several. This is because of the high energy capabilities connected with the desire to unify the design for MSR mission (possibility to deliver ~120 kg ERC from Mars orbit to the Earth). The principle scheme of the separation system which is placed in ERC TPS area and the ERV design options are shown on Figure 7-1 and Figure 7-2 respectively.

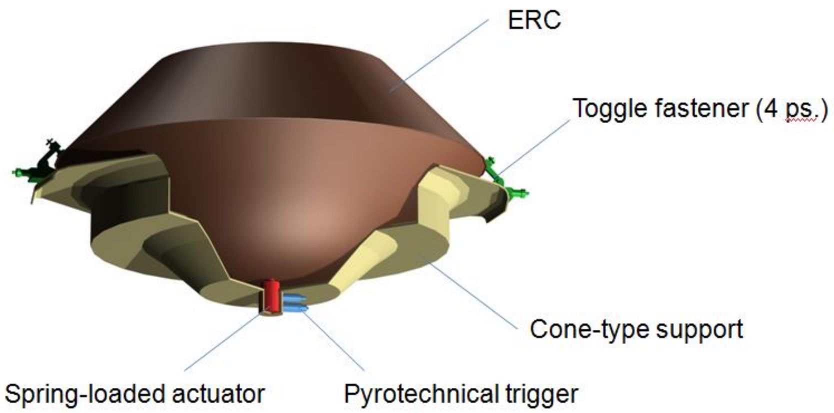


Figure 7-1: Principle scheme of the ERV/ERC separation system

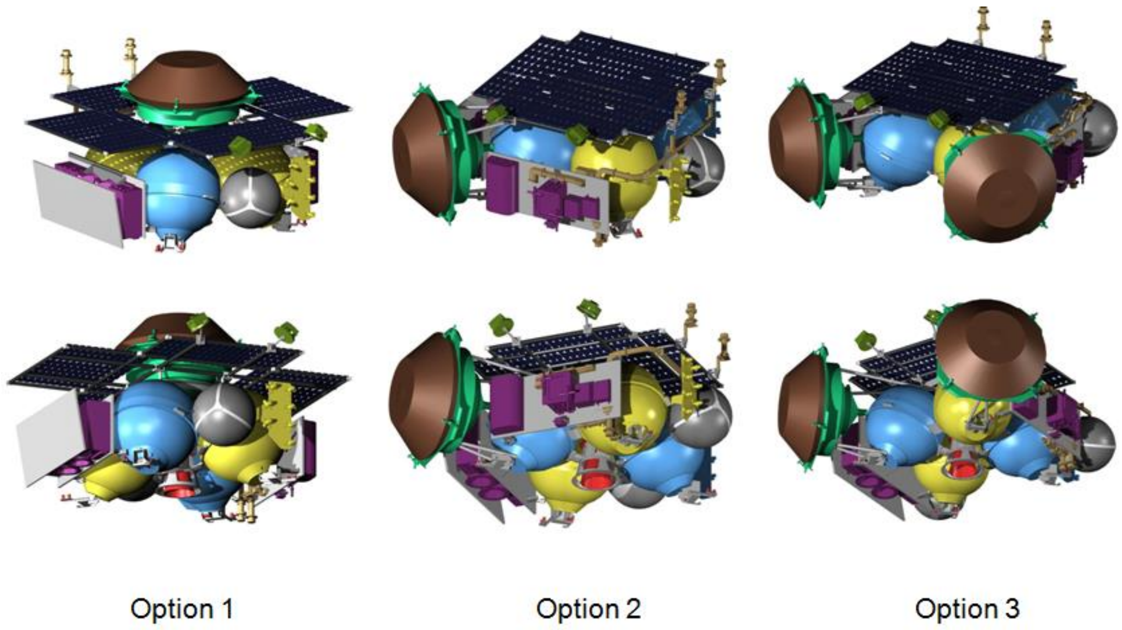


Figure 7-2: ERV design options with separation system placed in ERC TPS area

Considered design options include different methods of loading Phobos samples into the ERC. Sample loading manipulator for options 2 and 3 is more lightweight and has simpler design than its analogue for option 1. Schematic diagram of samples loading process is shown on Figure 7-3.

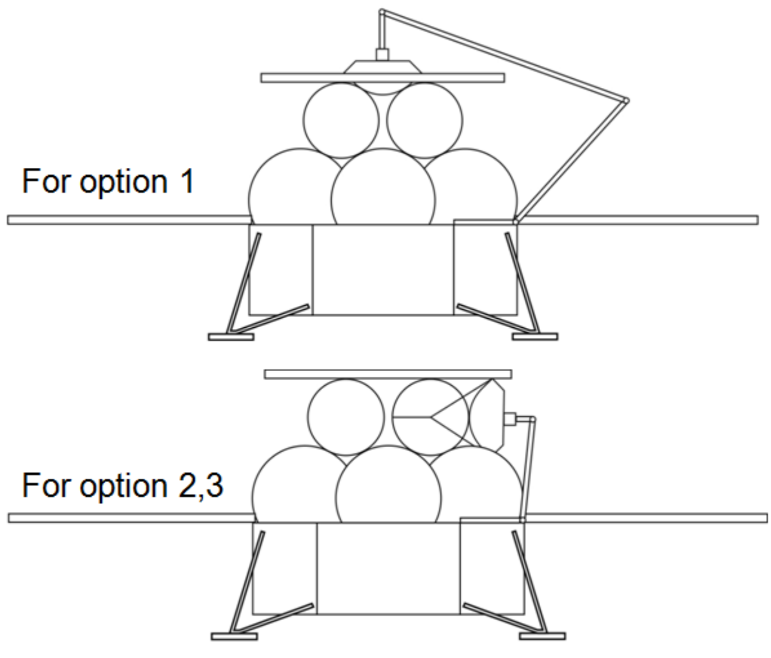


Figure 7-3: Schematic diagram of the samples loading process

During the options discussion, it was clarified that European ERC design does not allow mechanical impact of the spring-loaded actuator (a part of the separation system) on ERC TPS. This fact means that it is impossible to implement design options which are shown above. The decision was to move the ERC separation system under ESA responsibility and place it in ERC backshell area.

This agreed decision led to substantial changes to the design and system configuration solutions, namely:

- Changing the structural concept. In earlier design options, the main structure was composed of interconnected propellant tanks. The ERC with a cone-type support and the separation system was installed on the top of this structure with the backshell directed upwards. In the final design, the cone-type support is integrated in the main structure. The ERC is now installed inside and mounted in the backshell area. The propulsion system and other onboard equipment are mounted on this structure
- Increase of the installation radius of the propellant tanks due to the main structure geometry
- Changing the thrusters configuration, from one large central main engine to four smaller radial engines
- Equipment panels mounting principle modification.

Final design version would also allow placing the heavy-weight ERC (~120 kg) for MSR mission inside ERV. The general view of the final ERV design is shown on Figure 7-4.

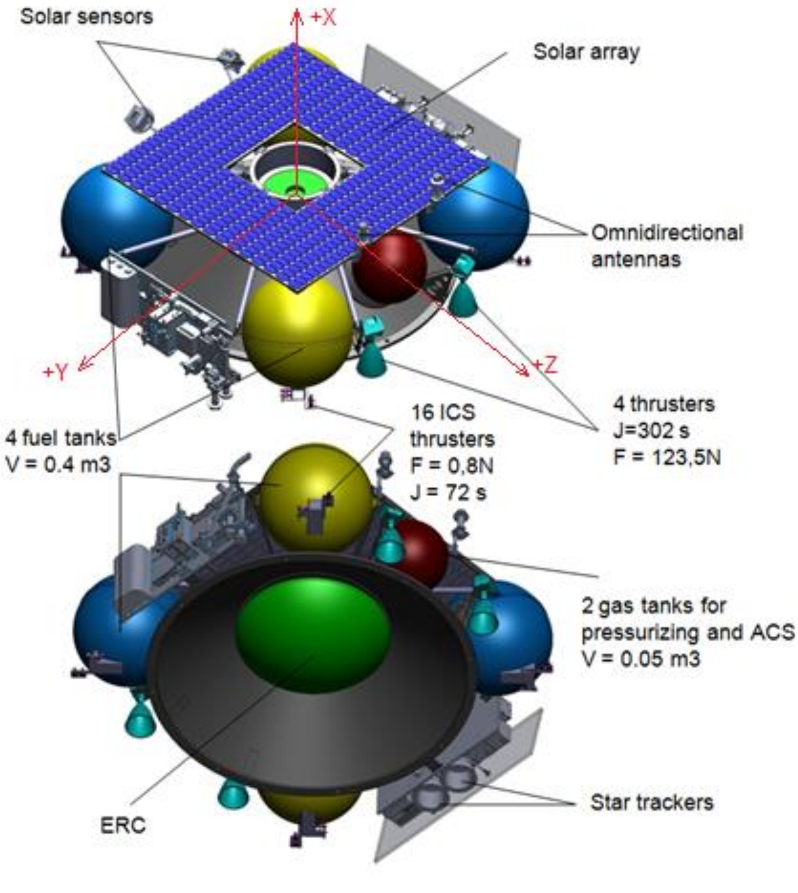


Figure 7-4: General view of the final ERV design

7.2 Design Description

The ERV is an autonomous SC the main purpose of which is to deliver ERC with samples from Phobos to the Earth. ERV has 2 propulsion systems:

- Bipropellant main propulsion system (NTO/N2H4) with pressurised type of propellant supply
- Monopropellant cold gas ACS propulsion system (N2).

The main propulsion system includes 4 engines with Isp=302s, T=123,5N, 2 NTO tanks and 2 N2H4 tanks. The overall volume of the tanks is V=0.4m^3. ACS propulsion system includes 16 cold gas thrusters with Isp=72s, T=0,8N and 2 tanks with overall volume V=0,05m^3 to provide orientation and stabilisation of the ERV and also for the pressurisation of the main propulsion system tanks. The final decision on the architecture and the characteristics of the propulsion system could be refined after more detailed analysis. On-board equipment panels with ACS and X-band transceiver are placed on ±Y axes. The EPS of the ERV includes the battery and the solar panel, which is oriented on +X. The main structural element of the ERV is a cone-type support with the ERC installed inside. As written above, the propulsion system and other on-board equipment are mounted on this structure.

The dry mass of the ERV is about 220 kg, propellant mass ~230 kg. Total mass of the ERV is 450 kg. Propellant tanks with joint volume V=0.4m^3 allows potential filling of up to ~450kg of propellant, which would be required in the MSR case (Figure 7-5).

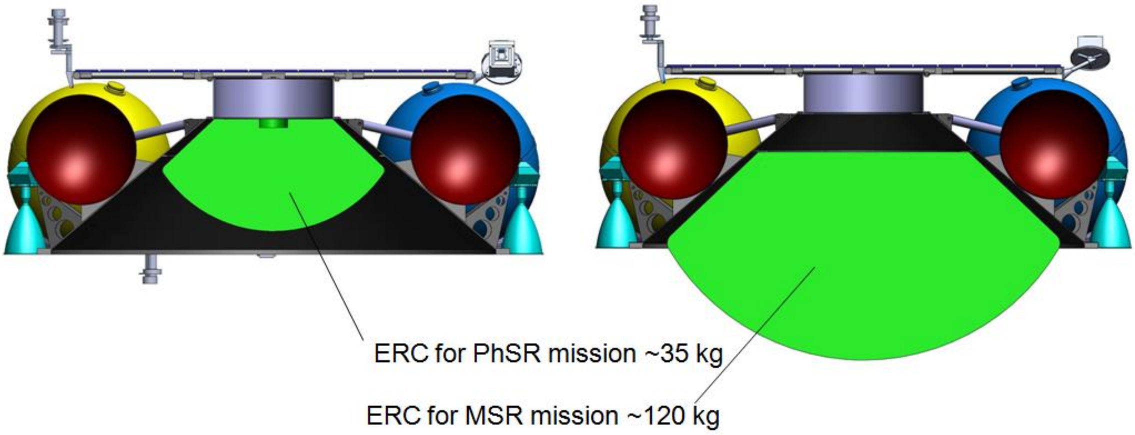


Figure 7-5: Unified cone-type support for heavy-weight ERC placement

To provide power supply for the ERV on-board equipment, it is proposed to use its own EPS on all mission phases, but it will be more preferable to have some additional power from LM for risk reduction in case of emergency.

According to NPOL assumptions, maximum level of power consumption by ERV on-board equipment shall not be more than 150 Watt.

7.3 Options

Combined propulsion system (N_2H_4/NTO+ N_2) might be replaced with a single bipropellant type propulsion system (N_2H_4/NTO) to increase reliability, decrease mass and unify the propulsion system with systems which have been already manufactured and tested.

7.4 Interface Requirements

During the joint CDF study, it was agreed that the Parties shall have as few electrical and data interfaces between SC modules as possible. As for the ERV EPS, it is necessary to provide solar orientation before landing on Phobos to charge the ERV battery using the ERV solar array before landing on Phobos.

Regarding the mechanical interfaces between the SC modules, the Parties have agreed on preliminary mounting dimensions and also on responsibility sharing scenario for the adapters and separation systems. The adapter and the separation system between LM and ERV are under ROSCOSMOS responsibility, the general view of the adapter and its mounting dimensions are shown in Figure 7-6.

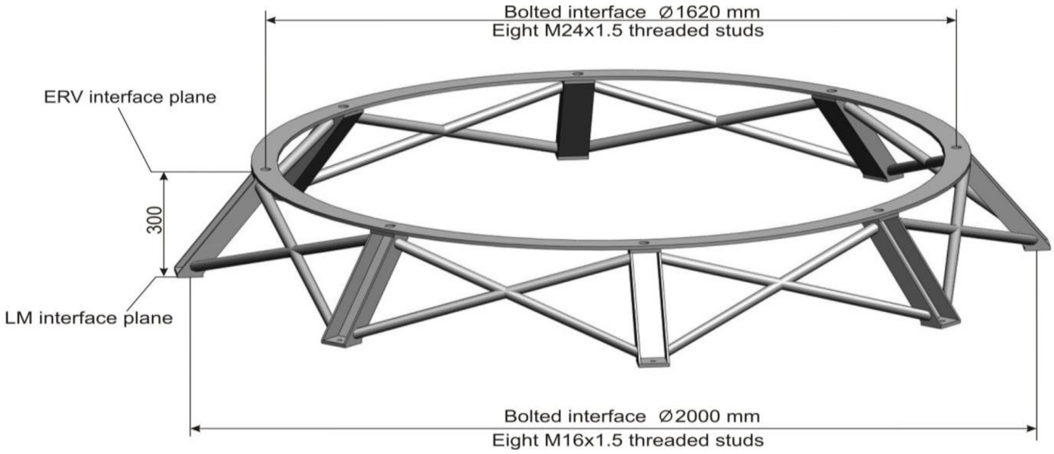


Figure 7-6: Proposal for LM/ERV adapter

The adapter with the separation system between ERV and ERC is under ESA responsibility, but the command for separation shall be sent from ERV OBC, which is under ROSCOSMOS responsibility. The general scheme of ERV/ERC adapter placement is shown in Figure 7-7.

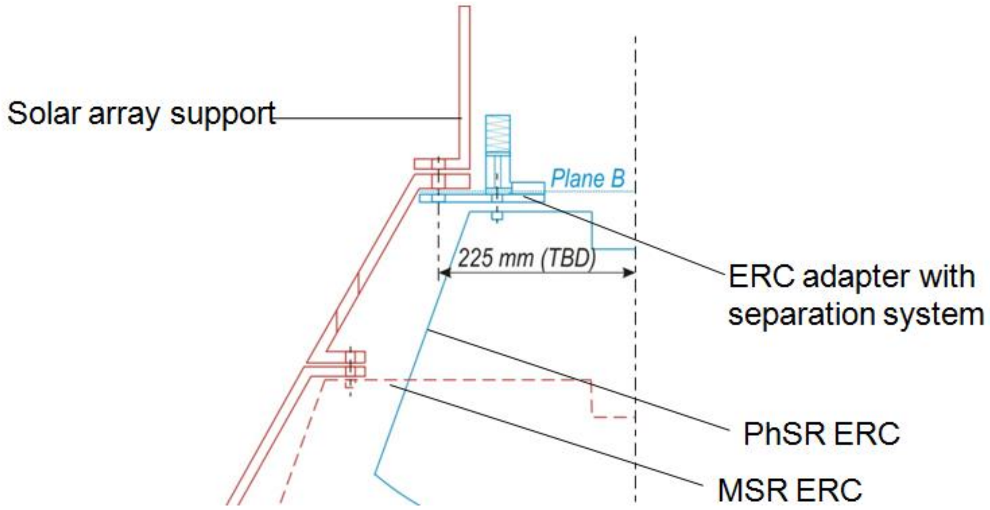


Figure 7-7: Proposal for ERV/ERC placement

7.5 Technology Requirements

Technology readiness level of ERV systems and its components are shown in Table 7-1.

System		TRL status
Propulsion system		
1	**Propulsion system**	7
1.1	Main propulsion engines	8
1.2	ACS propulsion engines	8
1.3	Propellant storage and transfer system	7
1.4	Gas storage and transfer system for pressurising and providing orientation and stabilisation	7
Power supply system		
2	**Power supply system**	5
2.1	Storage battery	5
2.2	Solar panel	5
2.3	Power conditioning unit	5
Onboard Control Complex		
3	**OCC**	7

System		TRL status
3.1	OBC	8
3.2	IMU	6
3.3	Star tracker	9
3.4	Solar sensor	9
3.5	Control Unit	7
3.6	Control algorithms, software	5
4	**X-band transceiver**	6
5	**TCS**	7
6	**Separation systems**	9
7	**Structure**	4

Table 7-1: ERV systems and components TRL

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8 SYSTEM

8.1 System Requirements and Design Drivers

The mission and system requirements were defined in advance of the Phobos Sample Return CDF study in agreement with the customer (D-SRE) and the Russian partners. Note that the system requirements were established for the propulsion module, the landing module, the Earth return vehicle, the Earth re-entry capsule as well as the complete composite and are indicated as PM, LM, EV, ER, and CO respectively.

Mission Requirements	
Req. ID	**STATEMENT**
MI - 010	The mission shall return approximately 100g of loose material from the surface of Phobos
MI - 020	The mission shall perform a series of science measurements of Deimos and Phobos using the payload as defined in chapter 5.
MI - 030	The mission shall be compatible with the science requirements defined in chapter 3.2, and with additional science objectives (following ESA/IKI discussions) defined in chapters 3.2 and 5.
MI - 040	The mission shall be designed for a launch in 2024 as a baseline, with 2026 as back up *C1: any type of transfer identified by mission analyses in this timeframe shall be checked* *C2: the requirement implies that the composite design shall be compliant with both the baseline and backup launch dates* *C3: mission compatibility with 2022 and 2028 launch dates shall also be checked for information*
MI - 050	The mission shall be launched by Proton-M from Baikonur in a direct escape trajectory. *C: the possible interest of an injection in an intermediate Earth orbit followed by a set of manoeuvres performed by the PM for escaping Earth, shall be checked*
MI - 060	The launch windows and transfers characteristics shall be as per chapter 4 of this report.
MI - 070	At Mars arrival, the mission shall perform a series of manoeuvres, as per chapter 4 of this report, in order: - First to allow for Deimos characterisation with the science payload, by reaching a QSO around Deimos - Then to reach its operational orbit (QSO) around Phobos *C: in case of non-compliance with the launch mass, fly-by around Deimos instead of QSO may be considered*
MI - 080	The Phobos science characterisation measurements shall be performed from three types of orbits: - A trailing orbit, at the end of the phasing phase, when the spacecraft is on an almost-Phobos-orbit and is getting closer to Phobos - A Quasi Satellite Orbit around Phobos, also called operational orbit, during which a pre-selection of landing sites is performed - Fly-bys orbits over the pre-selected sites for finalising the landing site selection *C: The Deimos observation strategy shall be less exhaustive than for Phobos, and will be defined during the study*

Mission Requirements	
Req. ID	**STATEMENT**
MI - 090	Contextual science information from the selected landing site shall be provided to the ground using: - Possibly images taken during the final descent to Phobos (the navigation camera may be used for this purpose) - Images of the sampling area taken once landed, from the dedicated context cameras described in Table 5-1
MI - 100	The trailing orbit shall place the spacecraft in a range of TBD to TBD km behind or ahead of Phobos
MI - 110	The Quasi Satellite Orbit shall have the following characteristics (TBC): - Altitude to Phobos between 50 and 100 km
MI - 120	The fly-by trajectories shall allow to pass over the potential landing sites at an altitude lower than 5 km (TBC), for high resolution science measurements. *C: the altitude of the fly-bys shall be consolidated considering science payload requirements as well as mission constraints*
MI - 130	A minimum of 3 (goal: 5) fly-bys shall be performed.
MI - 140	The mission shall allow to access 20% (goal: 50%) of the Phobos surface for the sampling operations.
MI - 150	The mission shall perform a series of manoeuvers for escaping Mars, as per RD[2]
MI - 160	The mission design shall be such that operations can take place from a single ground station during non-critical phases *C: For critical phases, the need of more than one ground station shall be justified*
MI - 170	The total duration of the Phobos far range characterisation phase on the trailing orbit shall be <100 days TBC
MI – 180	The total duration of the Phobos medium range characterisation phase on the QSO orbit shall be <100 days TBC
MI - 190	The total duration of the Phobos close range characterisation phase using a set of fly-bys shall be <25 days TBC
MI - 200	The mission shall perform a static landing (no touch-and-go) allowing for sampling operations with a stationary lander on the Phobos surface (TBC).
MI - 210	Once landed, the mission shall stay on the Phobos surface for a duration allowing to perform the sampling location selection, the sampling, transfer and sealing operations, as well as the post-sampling science measurements operations
MI - 220	The mission shall perform 3 types of surface operations: - Sampling point selection and characterisation - Sample acquisition and transfer to ERC - Post-sampling science measurements
MI - 230	The ERV shall lift-off in a safely manner as soon as possible after the sample transfer and ERC closing operations have been completed.
MI - 240	The mission shall allow for one landing on Phobos *C: provision for more landings would be delta-v consuming while the proposed landing and*

Mission Requirements	
Req. ID	**STATEMENT**
	sampling approach provide flexibility for managing sampling operations contingencies
MI - 250	The mission shall allow for: - One rehearsal of the landing sequence up to a TBD point above the landing site - One abort (either autonomous or from ground) of the landing sequence. An autonomous collision avoidance procedure shall be implemented
MI - 260	Once landed, the mission shall allow the Ground to select the sampling location within the sampling tool range
MI - 270	The mission shall provide the possibility to the Ground to check that the collected sample is suitable before transfer to the ERC
MI - 280	The mission should implement on-board automatic procedures to perform contingency sampling and lift-off operations in case of communication failures with the ground
MI - 290	No critical operation shall be performed if the Sun-Earth-Spacecraft angle is lower than 5 deg
MI - 300	No standard operation shall be performed if the Sun-Earth-Spacecraft angle is lower than 2 deg
MI - 310	The mission Planetary Protection category shall be: category V, unrestricted Earth return (TBC). *C: This requirement is pending confirmation by COSPAR*
MI - 320	For the outbound leg and the Mars orbital phase, the Planetary Protection category III requirements shall apply.
MI - 330	The LM operations duration on the Phobos surface (after ERV take-off) - to implement science measurements (eg USO, DIAMOND, LIBRATION,….) - will be > TBD

Table 8-1: Mission requirements

System requirements	
Req. ID	**STATEMENT**
CO-10	The difference between the launcher performance and the Composite wet mass including adapter shall be positive
CO-20	The Composite shall be compatible with the Proton launch environment.
CO-30	The Composite shall provide single point failure tolerance. Redundancy concepts shall be considered to minimise consequences of single point failures *C: any deviation with respect to this requirement shall be identified and justified*
CO-40	The lifetime of the Composite shall be compatible with the longest mission duration resulting from the mission trajectories selected, including contingencies
CO-50	The Composite design shall be compatible with the worst case delta-V among the selected mission launch windows and trajectories, including contingencies
CO-60	In the Composite design, only technologies that can be assumed to be at TRL 5 at the start of the mission implementation phase shall be considered when defining the mission

System requirements	
Req. ID	**STATEMENT**
	architecture
CO-70	The Composite design shall be compatible with the Planetary Protection requirements
CO-80	Throughout the Composite development phase, the possible contaminants to the sample shall be tracked and the parts to be in contact with the sample cleaned such that the contamination requirements in RD[4] are fulfilled
CO-90	Witness plates shall be used on-ground and in-flight to track possible contaminants to the sample
LM-10	The LM shall carry the ERV + ERC stack
LM-20	The LM shall accommodate the payload instruments suite as defined in Table 5-1 and Table 5-2 and provide proper interfaces.
LM-30	The Absolute Pointing Error (APE) at instruments interface shall be less than TBD deg
LM-40	The Relative Pointing Error (RPE) at instruments interface shall be less than TBD deg/s
LM-50	Science instruments shall not be used as baseline GNC sensors. *C: however science camera may be used for Orbit Determination in QSO if properly justified*
LM-60	The LM shall allow the sampling, transfer to ERC and sealing of the sample
LM-70	The sampling mechanism shall be placed at the tip of a robotic arm allowing to: - reach a a circle of 1,2 m radius and 170 degrees angle (TBC) transfer the sample to the ERC.
LM-80	The LM shall provide containment to the sample in order to fulfil the RD[4] contamination requirements
LM-90	The sample containment shall be able to sustain the ERC hard landing environment.
LM-100	The science payload and sampling tool shall allow to fulfil the science requirements as per RD[4]
LM-110	The LM shall be capable to perform all the required mission manoeuvres from the PM separation to the landing on Phobos
LM-120	The landing accuracy on Phobos shall be better than 50m (goal, 3-sigma) *C: In case it is found that this requirement cannot be fulfilled, possible relaxation shall be discussed with scientists*
LM-130	The LM shall be able to land successfully on a terrain having the following characteristics: - Slope (with respect to mean terrain horizontal) < 10 deg TBC Boulder size < 35 cm (goal 50cm) TBC
LM-140	The landing velocities at Phobos shall be as follows: - Vertical velocity < 100 cm/s TBC Horizontal velocity < 15 cm/s TBC
LM-150	The LM attitude excursion at landing with respect to the mean terrain horizontal shall be within -5deg and +5deg TBC.
LM-160	At landing, the LM shall minimise re-bounce.
LM-170	At landing the LM shall ensure that no part other than the landing gear shall touch the ground (including boulders), with a margin of TBD.

System requirements	
Req. ID	**STATEMENT**
LM-180	The landing phase shall end by a free-fall without using the thrusters, *C: the altitude of the free-fall shall be traded at system level considering several criteria such as the landing gear sizing, the landing stability, the soil contamination by the thrusters, the relative attitude of the spacecraft to the ground, etc… 20m can be taken as starting point.*
LM-190	For the landing phase the LM GNC system shall perform relative navigation vs Phobos, using a dedicated GNC camera and an altimeter. *C: absolute navigation may be implemented at ground level by introducing a way-point before the final descent allowing to check the position of the spacecraft wrt the target site.*
LM-200	The LM shall ensure the required stability during the sampling operations, using thrusters and/or an anchor
LM-210	The LM shall have its own power, communication and data handling subsystems
LM-220	The LM telecommunication systems shall be compatible with the ESA and ROSCOSMOS Deep Space Ground Stations
LM-230	The LM shall perform all communications in X-band *C: the antennas strategy is to be defined during the study*
LM-240	The communication and data handling subsystems shall be able to store and transmit to the ground all the necessary housekeeping and science information during all the phases of the mission, including contingency modes.
LM-250	The communication subsystem shall support the two-way Ranging and Doppler measurements of the Spacecraft throughout all mission phases and Delta DOR if high recision navigation is required.
LM-260	Real-time information shall be provided directly to Earth during the descent to Phobos, allowing the monitoring of the major events. *C: the level of information (carrier only, tones, data) shall be discussed during the study*
EV-10	The ERV shall carry the ERC to Phobos and back to Earth
EV-20	The ERV shall provide all necessary resources and interfaces to the ERC. *C: this concerns in particular the ERC beacon triggering and possibly electrical resources if deemed necessary*
EV-30	The ERV shall be capable to perform all the required mission manoeuvres from lift from Phobos to return to Earth, including mid-course corrections
EV-40	The ERV shall have its own power, communication and data handling subsystems
EV-50	The ERV telecommunication systems shall be compatible with the ESA and ROSCOSMOS Deep Space Ground Stations
EV-60	The ERV shall perform all communications in X-band *C: the antennas strategy is to be defined during the study*
EV-70	The communication and data handling subsystems shall be able to store and transmit to the ground all the necessary housekeeping and science information during all the phases of the mission, including contingency modes.
EV-80	The communication subsystem shall support the two-way Ranging and Doppler measurements of the Spacecraft throughout all mission phases and Delta DOR if high precision navigation is required.

System requirements	
Req. ID	**STATEMENT**
EV-90	The ERV shall release the ERC in a hyperbolic trajectory at arrival at Earth, with release errors lower than TBD, and with a spin rate of TBD for stabilisation purpose. *C: time of release before re-entry to be determined at system level, including necessary operations from ground. Typically between 1 to 4h.*
EV-100	The ERV shall perform an Earth avoidance manoeuver after the ERC release (TBC).
PM-10	The PM shall carry the LM from beginning of mission to PM separation
PM-20	The PM shall be capable to perform all the required mission manoeuvres from beginning of mission to PM separation
PM-30	The PM shall be commanded by the LM, and PM power shall be provided by the LM.
ER-10	The ERC shall safely land the sample container on Earth such that the sealing integrity is preserved
ER-20	The ERC design shall be compatible with the Earth entry conditions of the return legs selected from RD[2]
ER-30	The ERC shall perform a fully passive re-entry.
ER-40	The ERC should perform a night re-entry *C: for helping the optical tracking and the sample temperature containment*
ER-50	The ERC shall be spun by the Spacecraft at separation for stabilisation with a TBD spin rate
ER-60	The ERC landing site shall be in Kazakhstan (TBC)
ER-70	The ERC shall allow a recovery time of the sample container on Earth surface within 4 hours TBC
ER-80	The ERC design shall be such that it can be tracked from ground during descent
ER-90	The ERC design shall provide the capabilities for an RF beacon for 4 hours after landing (TBC)
ER-100	The ERC design shall not feature any parachute system and shall perform a free-fall descent down to landing
ER-110	The ERC design shall be such that the maximum impact deceleration of the sample container at ERC landing is < 2000g TBC (goal: 800g) quasi-static load
ER-120	The ERC design shall be such that the maximum temperature seen by the sample container is < 40°C TBC.
ER-130	The ERC design shall allow to recover the sample container without damaging the seal
ER-140	ERC should carry a small capsule (0,1 kg) with biomaterials (without any actions) (TBC)

Table 8-2: System requirements

8.2 System Assumptions and Trade-Offs

In preparation of the Phobos Sample Return CDF study, the cooperating partners have agreed on a baseline and a backup split of responsibilities (see Figure 8-1). For both of these scenarios, the following assumptions have been made:

- Launch into direct escape trajectory with Proton and Breeze-M from Baikonur

- Baseline launch date in 2024, backup launch scenario in 2026
- Re-entry of ERC in Kazakhstan and return of approximately 100g loose material from Phobos.

	Baseline		Backup	
	ESA	ROSCOSMOS	ESA	ROSCOSMOS
Launcher				
Transfer Propulsion Module (PM)				
Landing Spacecraft (LM)				
Sampling and Transfer Equipment		With ROSCOSMOS participation		With ROSCOSMOS participation
Earth Return Vehicle (ERV)				
Earth Re-Entry Capsule (ERC)				
Science Instruments				
Launch Ops				
SC Cruise + Landing Ops				
ERV SC Operations				
Science Operations				
Ground support				
Sample Receiving facility				
Science exploitation				

Figure 8-1: Baseline and backup ESA-ROSCOSMOS cooperation scenarios

8.2.1 Staging Analysis

A staging analysis has been performed to identify the optimum split of responsibilities among the different elements of the composite, in order to minimise the overall launch mass. The constraints hereafter apply in the context of the agreed baseline cooperation scenario:

- The ERC (ESA responsibility), with high heritage from past industrial and CDF studies, is fixed at 35 kg including margin
- The ERV (RUS responsibility) is inherited from Phobos-Grunt and has a dry mass of 220 kg. This module is responsible for the inbound transfer from the surface of Phobos and is not an object of the staging analysis.
- The LM (ESA) is a new design which is optimised for this mission
- The PM (RUS) is based on the Fregat upper stage, therefore its dry mass is roughly fixed and its propellant load is object of the optimisation. The maximum propellant capacity is 5000 kg.

The following figure shows the different staging options analysed, focusing mainly on the outbound transfer. Four main manoeuvres have been retained for this analysis and detailed in section 8.5.2.1: MOI (Mars Orbit Insertion), TOA 1 & 2 (Target Orbit Acquisition) at Deimos and the Deimos-Phobos transfer

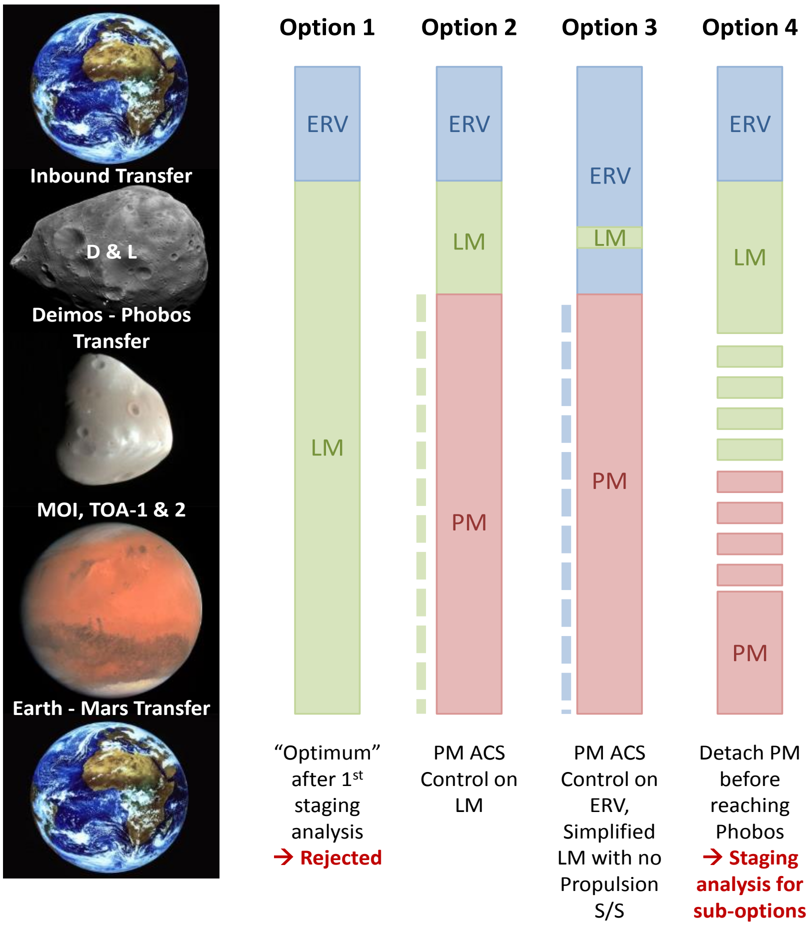


Figure 8-2: Staging options analysed

As shown before, four options have been identified:

- **Option 1** A large Landing Module responsible for all the outbound delta-v and no PM: This option resulted in a theoretical mass optimum (because the LM would be optimised on the exact delta-v) but would imply a very large LM at the boundaries of feasibility. In addition this solution would not respect the agreed cooperation scenario. *Option discarded*
- **Option 2** LM responsible for D&L and for the RCS of the overall composite: This implies that all the main delta-v are provided with the Russian PM, and allows to simplify the LM propulsion architecture as no high delta V manoeuvres are needed. This option is proved to be feasible but could be seen as a limit sub-case of option 4 and is therefore merged with it. *Option treated as sub-case of option 4*
- **Option 3** Simplified LM with no propulsion s/s. RCS and D&L provided by the ERV: This option implies a significant oversize of the ERV which - with the current design - does not have the sufficient control authority for the overall composite, especially in the early mission phases. *Option discarded*
- **Option 4** PM detached before reaching Phobos, LM responsible for RCS and remaining manoeuvres: This option is the most promising as it is possible to use

the existing PM based on Fregat design with minimum design update, as well as sizing appropriately the LM to control the whole composite. Many sub-options are possible and more detailed analyses are required. *Option baselined for further analysis*

8.2.1.1 Option 4 detailed staging analysis

A staging analysis has been performed for option 4, to identify the ideal split of manoeuvres for the outbound transfer between LM and PM. For the LM both a bipropellant and a monopropellant approach have been considered. The results are summarised in Table 8-3 (results are in the form of ratio to maximum launch mass).

	Low-Thrust manoeuvers	High Trust Engine Maneuvers					
		MOI	TOA-1 Deimos	TOA-2 Deimos	Deimos-Phobos	Total Stack Mass LM Biprop	Total Stack Mass LM Hydrazine
Delta-V [m/s]	510	963	293	410	840		
Scenario 4A	LM	PM	LM	LM	LM	0.87	1.12
Scenario 4B	LM	PM	PM	LM	LM	0.88	1.07
Scenario 4C	LM	PM	PM	PM	LM	0.89	1.02
Scenario 4D	LM	PM	PM	PM	PM	0.93	0.94

Table 8-3: Option 4 sub-staging scenarios

This preliminary analysis shows that all the options are feasible if considering a bipropellant propulsion subsystem on the lander (with a preference for jettisoning the PM as soon as possible) while for the monopropellant (hydrazine) configuration the only feasible option seems to be the option 4D (i.e. jettisoning the PM at Phobos).

Despite an overall mass penalty, this solution presents significant advantages:

- Simpler propulsion subsystem
- Compact LM (good for stability on Phobos)
- Lower contamination risk for the sampling area
- Maximum exploitation of the PM.

➔ *Scenario 4D with Hydrazine is selected as mission baseline and further analysed*

8.3 Mission System Architecture

The Phobos Sample Return mission consists of four elements: the propulsion module (PM), the landing module (LM), the Earth return vehicle (ERV), and the Earth re-entry capsule (ERC).

In the baseline mission scenario the following mission phases are foreseen:

- Launch and Direct Escape
- Transfer Earth-Mars
- Transfer to Deimos
- Deimos Close Proximity Phase
- Deimos-Phobos Transfer
- Phobos Close Proximity Phase
- Descent and Landing Phase
- Surface Operations Phase
- Ascent Phase

- Departure Phase
- Transfer Mars-Earth
- Re-entry Phase.

These mission phases are also illustrated in Figure 8-3. As an alternative to this mission scenario and if deemed necessary, the transfer to and stay at Deimos can be negotiated with respect to mass and/or time savings. These backup scenarios are explained in more detail in section 8.3.1.

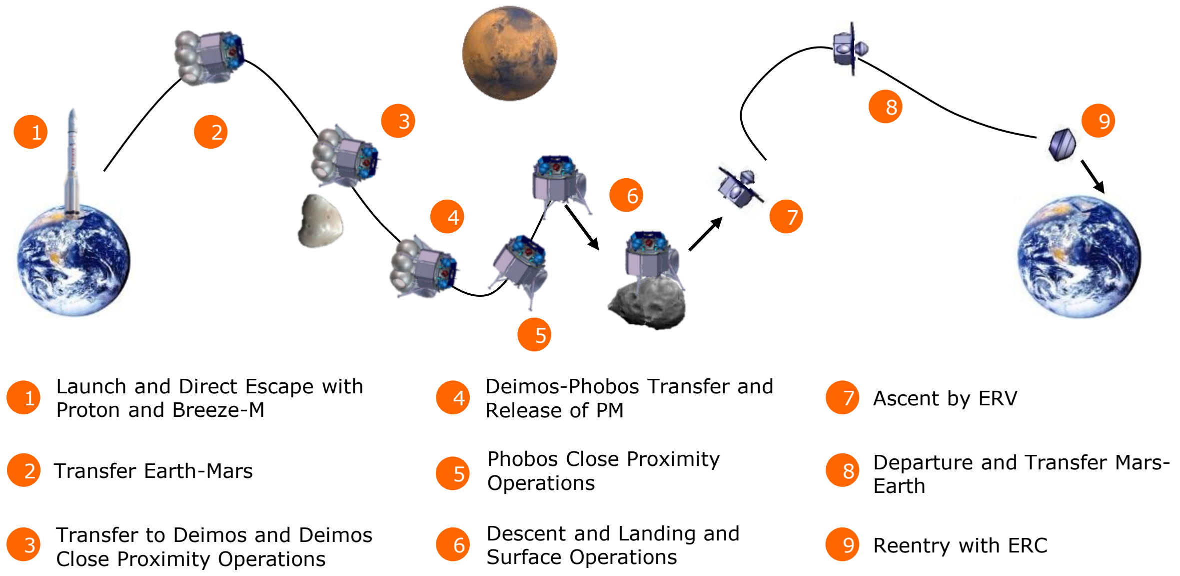


Figure 8-3: Mission Phases

Based on the mission phases as stated above, the mission timeline has been established during the CDF study by analysing short (less than 3 years) and long (less than 5 years) mission scenarios for the baseline as well as the backup launch date.

In Figure 8-4, the different mission scenarios are depicted for a launch in 2024 and 2026 respectively. It can be seen that to follow a short mission scenario, the time allocated for the scientific observations, i.e. QSO around Deimos and Phobos, needs to be constrained. However, at this early stage of the project this finding is seen as non-critical by the involved science support. Since the mission phase durations were derived from previous analyses of the MMSR-A5 and Phootprint studies, the time allocations can still be modified with regard to the scientific needs and mission feasibility. For the long mission scenarios, proposed allocations for the observation phase actually exceed significantly the currently envisaged duration. However this could allow for a more systematic remote sensing campaign, for instance with an improved coverage at different Solar illumination, less dependent on seasons, or with an improved 3D model build-up.

Note: Conjunctions (i.e. when Sun-Mars-Earth angle is below 5 deg) take place in January 2026, March 2028 and May 2030. Therefore for each short mission scenario one conjunction was taken into account while for each long mission scenario two conjunctions were considered.

	Launch 2024		Launch 2026	
Earth Departure Date	22/09/2024	12/10/2024	21/10/2026	10/11/2026
Arrival window size [d]	18		5	
Retained Mars Arrival Date (Worst case)	**09/09/2025**		**09/09/2027**	
Mars Departure date to Earth	R2026S 03/08/2026	R2028S 06/09/2028	R2028S 06/09/2028	R2030S1 09/11/2030
Available days (latest Mars arrival)	328	1093	363	1157
Mars Orbit Insertion	2	2	2	2
Target Orbit Acquisition (Deimos)	28	28	28	28
QSO around Deimos	25	300	40	365
Transfer to Phobos orbit	5	5	5	5
Trailing Orbit ahead of Phobos	20	100	40	100
QSO around Phobos	100	365	100	365
Fly-bys over selected landing site	25	25	25	25
Landing operations	14	14	14	14
Surface operations: sample point selection	3	3	3	3
Surface operations: sample acquisition & transfer	3	3	3	3
Ascent	2	2	2	2
Departure orbit acquisition	28	28	28	28
Trans Earth injection	2	2	2	2
Sun-Earth conjunction (operations not possible)	50	100	50	100
Margin w.r.t. departure transfer date	**21**	**116**	**21**	**115**
Arrival to Earth	19/06/2027	11/08/2029	11/08/2029	22/09/2031
Total Mission duration [d] (Worst case=LPO launch)	1000	1764	1025	1777
Total Mission duration [years]	*2.7*	*4.8*	*2.8*	*4.9*

Figure 8-4: Mission timeline depicting short and long mission scenarios for baseline and backup launch dates

8.3.1 Mission Options

A main mission alternative had been identified, to further increase the robustness of the baseline system design (e.g. to solve possible mass issues in the future): the de-scope (also in later stages) of the Deimos mission i.e. a complete de-scope of the science objectives related to Deimos OR a relaxation of them in order to allow the scientific observations of Deimos to be held from an intermediate elliptical orbit during the Phobos TOA phase, phased with Deimos. In this way it would be possible to minimise the required delta-v for the mission, at the expense of less exhaustive Deimos characterisation..

The mass saving in terms of propellant is obvious and will be detailed in section 8.5.4. The main advantage of this alternative is that it is an option that can be selected up to very late stages of the project. In fact even when the design of the overall composite will be finalised, in case mass problem would arise, it will be necessary only to update the mission operations concept and load a lower quantity of propellant in the tanks of the LM and PM, without any impact on the design of the modules.

8.4 System Modes

For the purpose of this study, the focus was put on the sizing modes needed to establish the power budget. Hence, the following system modes were derived:

#	Name	Description
1	Launch Mode	From lift-off to separation. Battery is fully charged. All subsystems are switched off, except for essential equipment.
2	Global Characterisation Phase (GCP) – Sun Mode	All subsystems are switched on (incl. navigation equipment, Tx and Rx), all scientific instruments are on but no thrusting. Spacecraft composite is illuminated by the Sun. Around Deimos PM is attached draining 300 W. Around Phobos PM is detached.
3	Global Characterisation Phase (GCP) – Eclipse Mode	All subsystems are switched on (incl. navigation equipment, Tx and Rx), few scientific instruments are on but no thrusting. Spacecraft composite is not illuminated by the Sun, therefore running on batteries. Around Deimos PM is attached draining 300 W. Around Phobos PM is detached. ERV is consuming 150W from its battery .
4	Local Characterisation Phase (LCP) – Sun Mode	All subsystems are switched on (incl. navigation equipment, Tx and Rx, and thrusting), all scientific instruments are on. Spacecraft composite is illuminated by the Sun. Around Deimos PM is attached draining 300 W. Around Phobos PM is detached.
5	Local Characterisation Phase (LCP) – Eclipse Mode	All subsystems are switched on (incl. navigation equipment, Tx and Rx, and thrusting), few scientific instruments are on. Spacecraft composite is not illuminated by the Sun, therefore running on batteries. Around Deimos PM is attached draining 300 W. Around Phobos PM is detached. ERV is consuming 150W from its battery .
6	Descent Mode	All subsystems are switched on (incl. complete AOCS suite, Tx and Rx). All scientific instruments are off. Split in power supply: two hours on solar array and one hour on batteries.
7	Surface Tx Earth Mode	Essential subsystems are switched on (no AOCS and propulsion but Tx continuously at full power), all scientific instruments are off. Spacecraft is on Phobos and illuminated by the Sun.
8	Surface Standby Night Mode	All subsystems are switched off, except for essential equipment (no Tx). All scientific instruments are switched off. Spacecraft is in the dark on Phobos, running on batteries. ERV is consuming 150 W from its battery .
9	Surface Day Operations Mode	All subsystems are switched off, except for essential equipment (no Tx). Cameras for surface operations and the sampling chain are switched on. Spacecraft is on Phobos and illuminated by the Sun.
10	Safe Mode	Hibernation and Failure Recovery mode: Instruments are put on standby or switched off. Non-essential functions are halted. TM/TC access to DHS is guaranteed to enable failure detection. Emergency Sun acquisition manoeuvre (not when landed).

8.5 System Baseline Design

The baseline design presented in the following sections is based, as anticipated, on 4 modules with a pre-agreed split of responsibilities between ESA and Russia. It is based on a decentralized architecture with maximum independence and interfaces as simple as possible between the Russian and the European modules.

The PM has no DHS and is controlled by the ESA LM, which provides also attitude control to the overall composite down to Phobos surface, and will be the module carrying the payloads for the scientific observations and the sampling chain for the sample collection and transfer in the ERC when on Phobos. The ERV will take-off from the surface and bring the ERC back to Earth where it will release it for re-entry. The LM instead, after ERV take-off, will remain active on the Phobos surface to continue the data downlink and possibly extend the science phase.

8.5.1 Overview

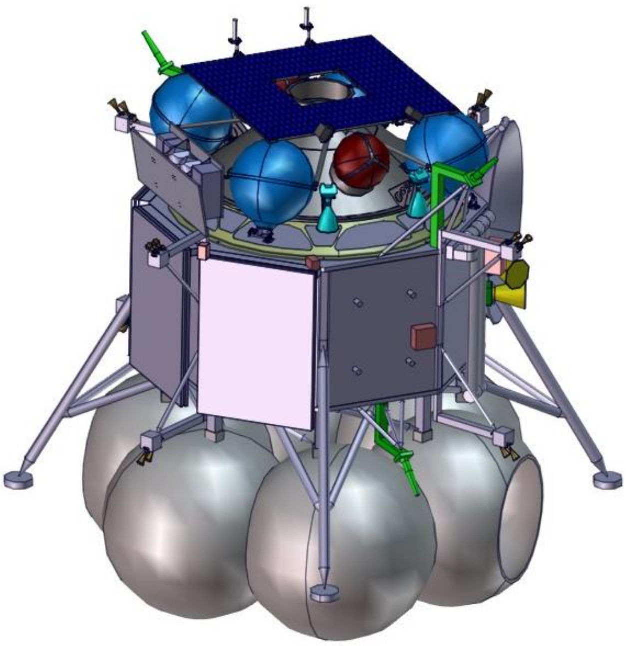


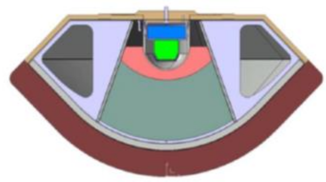
Figure 8-5 : Phobos Sample Return spacecraft composite

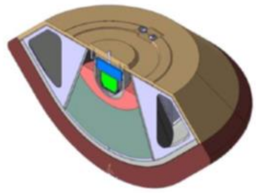
Composite Main Characteristics	
Mass (inc. Margin)	Dry Mass: 1694 kg Science Instruments Mass: 38.4 kg Max Propellant Mass: 3377 kg (launch 2026)
S/C Main Components	- ERC (Earth Re-entry Capsule) - ERV (Earth Return Vehicle) - LM incl. science P/L (Lander Module) - PM (Propulsion Module)

Table 8-4: Composite main characteristics

8.5.1.1 Earth Re-entry Capsule

Earth Re-entry Capsule Description		
Trajectory	Landing location	Kazakhstan
	Entry velocity	12.3 km/s (relative entry velocity - worst case retrograde)
	FPA	-9.8 deg (nominal)
	Mass	35 kg (incl. margin)
Shape	Scaled from Hayabusa 45° half cone front shield	
	Main Diameter	0.75 m
TPS	FS: ASTERM BS: Norcoat Liege	
	Heat load	Max: ¬ 221 MJ/m^2 (w. margin)
	Heat Flux	Max: ¬ 15 MW/m^2 (w. margin)
EDLS	None (no parachute)	
Structure	Load bearing Crushable materials to limit loads on sample	
Mechanisms	Sample container	
	Spin Separation device remaining on ERC	
GNC	None (uncontrolled re-entry)	
Communications	High g-load resistant recovery beacon based on aviation ELT or alternative	
DHS	None	





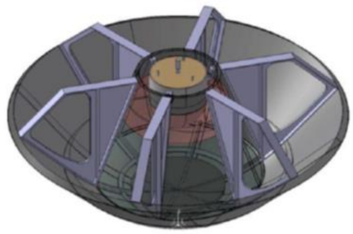


Table 8-5: Earth Re-entry Capsule main characteristics

8.5.1.2 Earth Return Vehicle

Earth Return Vehicle Characteristics		
AOCS/GNC	Sensors	Star Trackers
		Sun Sensors
		IMU
	RCS	16 x 0.8N Thrusters, cold gas
Propulsion	Bipropellant system, NTO/N2H4	
	Main Engine: 4 x 123.5N	
	Tanks: 4 fuel + 2 pressurant	
Power	SA	Body mounted
	Battery	1 x Lithium Ion
		BoL energy: 616 Wh
	On-board voltage	27±1.35V
Communications	All X-Band system	
	2 omni-directional antennas	
Thermal	MLI, heating lines, heaters	
DHS	OBC	
Mechanism	ERC spin separation device (SED TRP)	
	ERV separation remaining on ERV	
	Cable cutters	
	ERC ring hinge	
	ERC hold down	
Structure	Structural tanks + central cone for ERC accommodation	

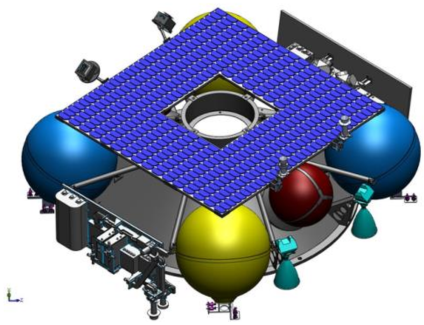


Table 8-6: Earth Return Vehicle main characteristics

8.5.1.3 Landing Module

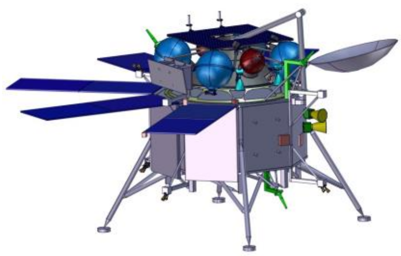
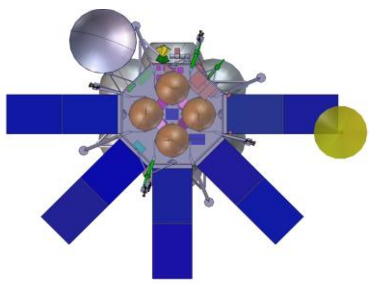
Lander Characteristics			
AOCS	D&L	Autonomous relative navigation	
	Sensors	2 x Star Tracker (AASTR)	
		2 x European IMU (Astrix 1090 + QA3000)	
		Wide Angle Cameras (2 OH) + (1EU), FoV: 53°	
		2 x Coarse Sun Sensor (TNO)	
		2 x Radar Altimeter	
	Actuator	4 x Reaction wheels (RSI 12/75-60)	
		16 / 24 x 20N thrusters	
Propulsion	Monopropellant system (Hydrazine)		
	Main engine: 1 x 1.1kN HTAE		
	Tanks	4 x Eurostar 2000 based, with 1801kg propellant	
Power	SA	5 x deployable wings	
		Solar cells: 30% 3J GaAs	
		Total area: 10.8 m²	
		1.2 kW (EOL Mars Orbit)	
	Battery	1 x Lithium Ion	
		BoL energy: 2600 Wh	
	Bus	28V MPPT regulated bus	
Communications	All X-Band system		
	1 x steerable HGA		
	3 x fixed LGA for 4π coverage		
	2 x TWT Power: 65W		
	2 x optional LGAs on PM		
Thermal	MLI, heating lines, Black Paint, SSM, insulating Stand-Offs		
	No heat pipes		
Mechanism	Sample	1 x Robotic arm incl. gripper	
		4 x landing legs	
		Sampling and containment tool (Rotary brushes)	
	Support	SA HDRM	
		HGA pointing mechanism	
		HGA pointing electronics	
		HGA resettable HDRM (RUAG)	
		Robotic arm HDRM	
		ERV separation device	
		ERV ejection springs	
DHS	OBC + MM based on LEON-FT		
Structure	Octagonal structure with CFRP and Al-Al panels. Corner beams transferring the load from the 8 PM hard points; top and bottom covers		

Table 8-7: Landing Module main characteristics

8.5.1.4 Propulsion Module

Propulsion Module Characteristics		
AOCS	None (Controlled by LM)	
Propulsion	Bipropellant system N2H2/NTO	
	Main engine: 20 kN	
	Tanks: 6 spherical	
Power	Chemical battery for propulsion power supply	
Communications	X-band antenna (Optional control by LM)	
Thermal	MLI, heaters	
DHS	None	
Structure	Structural tanks	

Table 8-8: Propulsion Module main characteristics

8.5.2 Budgets

Throughout the Phobos Sample Return CDF study delta-v and mass budgets have been analysed on different levels. In the subsections hereafter, these budgets are presented for the baseline as well as the backup launch date. Furthermore, the mass budget is also depicted for the baseline and backup cooperation scenario.

8.5.2.1 Delta V budget

The delta-v budget has been analysed for potential outbound transfers between 2022 and 2028 and the corresponding inbound transfers between 2026 and 2032. The resulting baseline and backup scenarios are depicted in Table 8-9 and Table 8-10.

For inbound as well as outbound transfers, the delta-v for the high thrust manoeuvres was provided by Mission Analysis while the low thrust manoeuvres are based on analyses that were conducted during the Phootprint phase A study. Latter contributes to sufficiently mature assumptions which were slightly adapted for this CDF study. On top of the provided delta-v, a 5% margin was applied to the high thrust manoeuvres and to the low thrust manoeuvres related to orbit change, while 100% margin was applied to AOCS manoeuvres as commonly applied in CDF studies (e.g. attitude control and RW offloading).

Outbound Transfers			P24S, LPC			P26S, LPC		
	Launch Date				12/10/2024			10/11/2026
	Arrival Date (before MOI)				09/09/2025			09/09/2027
Mission Phases	**Manoeuvre**	**Engine**	**DV [m/s]**	**Margin [%]**	**Total DV [m/s]**	**DV [m/s]**	**Margin [%]**	**Total DV [m/s]**
Transfer Earth-Mars (TEM)	After Launch Corrections	Low Thrust	10.00	5.00	10.50	10.00	5.00	10.50
	Deep Space Manoeuvre	High Thrust	0.00	5.00	0.00	0.00	5.00	0.00
	Correction when approaching Mars	Low Thrust	20.00	5.00	21.00	20.00	5.00	21.00
	Offset targeting	Low Thrust	40.00	5.00	42.00	40.00	5.00	42.00
	Total Delta-V TEM w/o GL				**73.50**			**73.50**
Transfer to Deimos	MOI impulsive	High Thrust	823.00	5.00	864.15	917.00	5.00	962.85
	TOA-1	High Thrust	262.00	5.00	275.10	279.00	5.00	292.95
	TOA-2	High Thrust	380.00	5.00	399.00	391.00	5.00	410.55
	Total Delta-V w/o GL				**1538.25**			**1666.35**
Deimos Close Proximity	Safe Mode	Low Thrust	1.00	100.00	2.00	1.00	100.00	2.00
	Attitude Control	Low Thrust	2.00	100.00	4.00	2.00	100.00	4.00
	Trailing -> Heading Orbit	Low Thrust	6.00	5.00	6.30	6.00	5.00	6.30
	QSO	Low Thrust	20.00	5.00	21.00	20.00	5.00	21.00
	Station Keeping	Low Thrust	6.00	5.00	6.30	6.00	5.00	6.30
	RW Off-loading	Low Thrust	2.00	100.00	4.00	2.00	100.00	4.00
	Provision for tilted thrusters	Low Thrust	8.25	5.00	8.66	8.25	5.00	8.66
	Total Delta-V PCP				**52.26**			**52.26**
Deimos-Phobos Transfer	Hohmann Transfer	High Thrust	800.00	5.00	840.00	800.00	5.00	840.00
	Total Delta-V DPT w/o GL				**840.00**			**840.00**
Phobos Close Proximity	Safe Mode	Low Thrust	1.00	100.00	2.00	1.00	100.00	2.00
	Attitude Control	Low Thrust	2.00	100.00	4.00	2.00	100.00	4.00
	Trailing -> Heading Orbit	Low Thrust	6.00	5.00	6.30	6.00	5.00	6.30
	QSO	Low Thrust	20.00	5.00	21.00	20.00	5.00	21.00
	High Latitude Observation	Low Thrust	35.00	5.00	36.75	35.00	5.00	36.75
	Movement to Safe Orbit	Low Thrust	20.00	5.00	21.00	20.00	5.00	21.00
	Station Keeping	Low Thrust	12.00	5.00	12.60	12.00	5.00	12.60
	RW Off-loading	Low Thrust	2.00	100.00	4.00	2.00	100.00	4.00
	Fly-bys (3)	Low Thrust	40.00	5.00	42.00	40.00	5.00	42.00
	Provision for tilted thrusters	Low Thrust	33.50	5.00	35.18	33.50	5.00	35.18
	Total Delta-V PCP				**184.83**			**184.83**
Descent and Landing	Direct landing, thrusters 45° tilted	Low Thrust	150.00	10.00	165.00	150.00	10.00	165.00
	Total Delta-V D&L				**165.00**			**165.00**
Surface	**Sampling / Hold-down Force (*)**	Low Thrust			0.00			0.00
	Total Delta-V Surface (*)				**0.00**			**0.00**
Total Delta-V Deimos + Phobos with GL					**2854**			**2982**

Table 8-9: Delta-v budgets for baseline and backup outbound transfer

The following influencing factors led to the delta-v budgets for the outbound transfer:

- No deep space manoeuvre is required during the transfer from Earth to Mars
- Around Deimos and Phobos the spacecraft composite enters into a quasi satellite orbit (QSO)
- Neither high latitude observations nor fly-bys are required around Deimos
- A movement from QSO to a safe orbit is foreseen during Sun-Earth conjunctions. Considering the assumed timeline this will happen only during the Phobos QSO, for a period of about 50 days.
- A Hohmann transfer is followed to transfer the spacecraft composite from Deimos to Phobos
- For the landing approach, a direct landing is assumed with the thrusters being 45 degrees tilted (the corresponding efficiency factor is included in the presented figures). The delta-v for landing includes an allocation for rehearsal and a second landing attempt

- An efficiency reduction of 25 % is assumed due to the 45 degree tilt of the thrusters for all the manoeuvres not related to attitude control, e.g. for the Deimos and Phobos proximity phases
- There is no delta-v allocated for the hold-down force during sampling but the required propellant mass is accounted for in the propellant budget in section 14.3
- Gravity losses are included in the margin taking into account that the Russian propulsion module features a 20 kN engine.

For the inbound transfer, the assumptions as stated below were the main drivers for the delta-v budget:

- The ascent of the ERV together with the ERC is initiated by springs. The ejection is followed by an ascent manoeuvre for which a delta-v of 20 m/s plus 20% margin is assumed
- For the TEI impulsive manoeuvre, gravity losses are explicitly accounted for as being 1% of the corresponding high thrust manoeuvre
- For a potential Earth gravity assist (EGA) a delta-v of 15 m/s plus 5% margin is assumed but would need to be discussed in a follow-up study in coordination with the responsible party on the Russian side for the purpose of the heading angle. Such an EGA would allow to choose the re-entry heading angle with almost no impact on delta-v.

	Inbound Transfers		R2026S			R2028S			R2030S1		
	Mars Escape Date				03/08/2026			06/09/2028			09/11/2030
	Earth Arrival Date				19/06/2027			11/08/2029			22/09/2031
	Duration				320			339			317
Mission Phases	**Manoeuvre**	**Engine**	**DV [m/s]**	**Margin [%]**	**Total DV [m/s]**	**DV [m/s]**	**Margin [%]**	**Total DV [m/s]**	**DV [m/s]**	**Margin [%]**	**Total DV [m/s]**
Ascent	Ascent	Low Thrust	20.00	20.00	24.00	20.00	20.00	24.00	20.00	20.00	24.00
	Re-orbitation to stable orbit pre DOA	Low Thrust	10.00	5.00	10.50	10.00	5.00	10.50	10.00	5.00	10.50
	Total Ascent				**34.50**			**34.50**			**34.50**
Departure (DOA & TEI)	DOA 1	High Thrust	743.00	5.00	780.15	743.00	5.00	780.15	743.00	5.00	780.15
	DOA 2	High Thrust	93.00	5.00	97.65	90.00	5.00	94.50	99.00	5.00	103.95
	TEI impulsive	High Thrust	799.00	5.00	838.95	704.00	5.00	739.20	636.00	5.00	667.80
	TEI impulsive GL		7.99	5.00	8.39	7.04	5.00	7.39	6.36	5.00	6.68
	Total Departure w/o GL				**1716.75**			**1613.85**			**1551.90**
	Total Departure with GL				**1725.14**			**1621.24**			**1558.58**
Transfer Mars-Earth	Departure Navigation	High Thrust	20.00	5.00	21.00	20.00	5.00	21.00	20.00	5.00	21.00
	DSM	High Thrust	0.00	5.00	0.00	0.00	5.00	0.00	0.00	5.00	0.00
	Approach Navigation	High Thrust	10.00	5.00	10.50	10.00	5.00	10.50	10.00	5.00	10.50
	EGA (TBC)	High Thrust	15.00	5.00	15.75	15.00	5.00	15.75	15.00	5.00	15.75
	Earth Re-entry Trajectory	High Thrust	15.00	5.00	15.75	15.00	5.00	15.75	15.00	5.00	15.75
	Solar Orbit Acquisition	High Thrust	10.00	5.00	10.50	10.00	5.00	10.50	10.00	5.00	10.50
	Total Delta-V TME w/o GL				**73.50**			**73.50**			**73.50**
	Total Delta-V TME with GL				**73.50**			**73.50**			**73.50**
Total Delta-V w/o GL					**1825**			**1722**			**1660**
Total Delta-V with GL					**1833**			**1729**			**1667**
	Launch 2024 (P24S, LPC)				Baseline			Backup			
	Launch 2026 (P26S, LPC)							Baseline			Backup
	Total Delta-V High Thrust Engine				1790			1687			1625

Table 8-10: Delta-v budget for baseline and backup inbound transfers depending on the launch date

8.5.2.2 Element mass budgets

The mass budget for the landing module is depicted in Table 8-11.

Lander						
	Without Margin	Margin		Total	% of Total	
Dry mass contributions		%	kg	kg		
Structure	121.47 kg	18.90	22.95	144.42	24.03	
Thermal Control	10.00 kg	20.00	2.00	12.00	2.00	
Mechanisms	83.25 kg	13.09	10.90	94.15	15.67	
Communications	25.10 kg	7.59	1.91	27.01	4.49	
Data Handling	11.00 kg	10.00	1.10	12.10	2.01	
GNC	36.30 kg	12.02	4.37	40.67	6.77	
Propulsion	93.46 kg	16.88	15.77	109.23	18.18	
Power	85.20 kg	15.07	12.84	98.04	16.32	
Harness	24.89 kg	0.00	0.00	24.89	4.14	
Instruments	32.00 kg	20.00	6.40	38.40	6.39	
Total Dry(excl.adapter)	522.66			600.90		kg
System margin (excl.adapter)		20.00	%	120.18		kg
Total Dry with margin (excl.adapter)				721.08		kg
Propellant	519.00 kg	N.A.	N.A.	519.00	41.85	
Total wet mass (excl.adapter)				1240.08		kg

Table 8-11: Mass budget for Landing Module

8.5.2.3 Composite mass budget

The mass budgets for the entire composite assuming a launch in 2026 and 2024 are presented in Table 8-12 and Table 8-13 respectively. It can be seen that for either option the margin with respect to the target launch mass is above 130 kg. Note that for the ERC an allocation of 35 kg including all margins has been used, based on Phootprint studies outcome, and that the PM does not consider system margin as being a quasi-direct re-use of Fregat.

Phobos Sample Return Composite - Baseline Option - Launch 2026					
	Mass w/o Margin [kg]	Margin [%]	Margin [kg]	Total Mass [kg]	
Dry mass contributions					
Propulsion Module incl I/F PM-LM	**674.0**	**0.0**	0.0	**674.0**	
Lander	**600.9**	**20.0**	120.2	**721.1**	
Earth Return Vehicle	**220.0**	**20.0**	44.0	**264.0**	
Earth Re-entry Capsule	**35.0**	**0.0**	0.0	**35.0**	
Total Dry with Margin				**1694.1**	**kg**
Wet mass contributions					
PM Propellant	**2628.0**	**0.0**	0.0	**2628.0**	
LM Propellant	**519.0**	**0.0**	0.0	**519.0**	
ERV Propellant	**230.0**	**0.0**	0.0	**230.0**	
ERC Propellant	**N/A**	**N/A**	-	**-**	
Total Wet Mass				**5071.1**	**kg**
I/F Breeze - PM	**100.0**	**0.0**	0.0	**100.0**	
I/F PM - LM (incl in Dry Mass calculation)	**N/A**	**N/A**	-	**-**	
I/F LM - ERV (incl. in LM structure mass)	**N/A**	**N/A**	-	**-**	
Launch mass				**5171.1**	**kg**
Target Launch Mass				**5302.0**	**kg**
Below Mass Target by:				**130.9**	**kg**

Table 8-12: Mass budget for composite assuming launch in 2026

Phobos Sample Return Composite - Baseline Option - Launch 2024					
	Mass w/o Margin [kg]	Margin [%]	Margin [kg]	Total Mass [kg]	
Dry mass contributions					
Propulsion Module incl I/F PM-LM	**674.0**	**0.0**	0.0	**674.0**	
Lander	**600.9**	**20.0**	120.2	**721.1**	
Earth Return Vehicle	**220.0**	**20.0**	44.0	**264.0**	
Earth Re-entry Capsule	**35.0**	**0.0**	0.0	**35.0**	
Total Dry with Margin				**1694.1**	**kg**
Wet mass contributions					
PM Propellant	**2436.0**	**0.0**	0.0	**2436.0**	
LM Propellant	**519.0**	**0.0**	0.0	**519.0**	
ERV Propellant	**230.0**	**0.0**	0.0	**230.0**	
ERC Propellant	**N/A**	**N/A**	-	**-**	
Total Wet Mass				**4879.1**	**kg**
I/F Breeze - PM	**100.0**	**0.0**	0.0	**100.0**	
I/F PM - LM (incl in Dry Mass calculation)	**N/A**	**N/A**	-	**-**	
I/F LM - ERV (incl. in LM structure mass)	**N/A**	**N/A**	-	**-**	
Launch mass				**4979.1**	**kg**
Target Launch Mass				**5130.0**	**kg**
Below Mass Target by:				**150.9**	**kg**

Table 8-13: Mass budget for composite assuming launch in 2024

8.5.2.4 Power budget

The power budget for the Landing Module is depicted in Table 8-14. Only the cases which have been found to be sizing for the Power Subsystem are summarised. The power required from the LM by the PM is included as well.

MISSION PHASE	LAUNCH	GCP - sun	GCP - eclipse	LCP - sun	LCP - eclipse	Descent	TxEarth	Night (Standby)	Day
LM	497	984	1001	675	758	696	471	256	322
Losses (LCL, harness)	15.1	35.0	41.5	28.8	38.5	29.8	18.1	13.6	11.1
OBDH S/S	36	55	55	55	55	55	35	31	31
EPS S/S	24	21	29	21	29	21	21	21	21
COMMS S/S	39	158	158	158	158	145	171	0	0
Thermal S/S	0	115	115	115	115	115	150	150	150
AOCS	0	103	103	108	161	161	0	0	0
MECHANISM S/S	0	0	0	0	0	0	0	0	39
PROPULSION S/S	3	3	3	45	45	58	0	0	0
PAYLOAD	0	36	36	36	36	0	0	0	18
PM	300	300	300	0	0	0	0	0	0

Table 8-14: Power budget

Note that it is assumed that the ERV is self-standing in power during the complete mission due to the orientation of its panels and the size of its batteries. Only emergency power supply is foreseen from the LM to the ERV. This is confirmed in the ERV description section.

8.5.3 Equipment List

The equipment list for the landing module is provided in Table 8-15.

Element 1 - Lander						
FUNCTIONAL SUBSYSTEM	nr	Mass (kg) per unit	Total Mass (kg)	Margin (%)	Margin (kg)	Mass (kg) with Margin
Structure			**121.47**	**18.90**	**22.95**	**144.42**
Panel lateral (w/o radiators)	6	3.49	20.93	20.00	4.19	25.12
Panel lateral (w/ radiators)	2	9.75	19.50	20.00	3.90	23.40
Panel top	1	14.83	14.83	20.00	2.97	17.79
Panel bottom	1	14.83	14.83	20.00	2.97	17.79
Strut/connector longitudinal	8	2.11	16.84	20.00	3.37	20.21
IF ring to ERV (octagonal)	1	3.20	3.20	10.00	0.32	3.52
IF ring to PM (octagonal)	1	10.20	10.20	10.00	1.02	11.22
Struts tank (4)	16	0.24	3.84	20.00	0.77	4.61
Fitting tank (1)	4	0.20	0.80	20.00	0.16	0.96
Brackets/connectors	1	13.50	13.50	20.00	2.70	16.20
Adapter top to ERV	1	3.00	3.00	20.00	0.60	3.60
Thermal Control			**10.00**	**20.00**	**2.00**	**12.00**
Overall_TCS	1	10.00	10.00	20.00	2.00	12.00
Mechanisms			**83.25**	**13.09**	**10.90**	**94.15**
Landing gears	4	5.60	22.40	20.00	4.48	26.88
Sampling tool	1	4.10	4.10	20.00	0.82	4.92
APM+APME+HDRM	1	12.50	12.50	10.00	1.25	13.75
SA deployment mech. + HDRM	6	5.00	30.00	5.00	1.50	31.50
Robotic arm	1	14.25	14.25	20.00	2.85	17.10
Communications			**25.10**	**7.59**	**1.91**	**27.01**
Transponder	2	3.50	7.00	5.00	0.35	7.35
LGA	3	0.30	0.90	5.00	0.05	0.95
TWTA	2	2.10	4.20	5.00	0.21	4.41
RFDN	1	3.00	3.00	10.00	0.30	3.30
HGA	1	10.00	10.00	10.00	1.00	11.00
Data Handling			**11.00**	**10.00**	**1.10**	**12.10**
OBC + MM	1	11.00	11.00	10.00	1.10	12.10
GNC			**36.30**	**12.02**	**4.37**	**40.67**
AA STR	2	1.50	3.00	20.00	0.60	3.60
European IMU (Astrix1090 + QA3000)	2	3.00	6.00	20.00	1.20	7.20
RW RSI 12/75-60	4	4.80	19.20	5.00	0.96	20.16
Radar Altimeter	2	2.00	4.00	20.00	0.80	4.80
Sun Sensor TNO	2	0.05	0.10	5.00	0.01	0.11
Wide Angle Camera (2OH+1EU)	2	2.00	4.00	20.00	0.80	4.80
Propulsion			**93.46**	**16.88**	**15.77**	**109.23**
20N thruster	24	0.38	9.12	5.00	0.46	9.58
Propellant tank	4	18.50	74.00	20.00	14.80	88.80
Propellant filter	1	0.30	0.30	5.00	0.02	0.32
Latching valve	2	0.70	1.40	5.00	0.07	1.47
Pressure transducer	3	0.28	0.84	5.00	0.04	0.88
Fill and Drain valve / Vent valve (propellant)	1	0.07	0.07	5.00	0.00	0.07
Fill and Drain valve / Vent valve (pressurant)	4	0.07	0.28	5.00	0.01	0.29
Piping (incl fittings)	1	2.00	2.00	5.00	0.10	2.10
Stand-off	1	2.50	2.50	5.00	0.13	2.63
Mounting screws	1	2.00	2.00	5.00	0.10	2.10
Miscellaneous	1	0.25	0.25	5.00	0.01	0.26
Pressurant	1	0.70	0.70	5.00	0.04	0.74
Power			**85.20**	**15.07**	**12.84**	**98.04**
Battery	1	25.00	25.00	10.00	2.50	27.50
PCDU	1	17.00	17.00	10.00	1.70	18.70
Solar Array	1	43.20	43.20	20.00	8.64	51.84
Instruments			**32.00**	**20.00**	**6.40**	**38.40**
WAC	*1*		*2.15*	*20.00*	*0.43*	*2.58*
NAC	*1*		*7.00*	*20.00*	*1.40*	*8.40*
NAC	1	7.00	7.00	20.00	1.40	8.40
baffle	1	0.00	0.00	5.00	0.00	0.00
CSU	*1*		*3.50*	*20.00*	*0.70*	*4.20*
Stereo Cam	*1*		*1.50*	*20.00*	*0.30*	*1.80*
CLUPI	*1*		*0.70*	*20.00*	*0.14*	*0.84*
IME	*0*		*0.00*	*0.00*	*0.00*	*0.00*
VisNIR	*1*		*7.10*	*20.00*	*1.42*	*8.52*
midIR	*1*		*6.30*	*20.00*	*1.26*	*7.56*
ADRON-RM	*1*		*1.60*	*20.00*	*0.32*	*1.92*
DIAMOND	*1*		*1.90*	*20.00*	*0.38*	*2.28*
LIBRATION	*1*		*0.25*	*20.00*	*0.05*	*0.30*
Propellant						**519.00**

Table 8-15: Equipment list for landing module

8.5.4 Mission Option Budgets

As anticipated in section 8.3.1, an assessment of the mass savings in case of de-scoping the Deimos mission has been performed. Since the Hohmann transfer from Phobos to Deimos is one of the last outbound manoeuvres of the composite, a significant reduction of that delta-v would have an immense effect on the overall propellant load, also on the LM.

The following table shows the updated mass budgets for the best saving case scenario, i.e. the complete dismissal of the Deimos mission, considering in this case longer operations at Phobos somehow equivalent to the ones at Deimos that have been descoped. (Meaning that only the complete Hohmann transfer has been removed from the delta-v calculation).

The expected mass saving is more than **1 t** leading to launch margins in the order of 20-25%.

Phobos Sample Return Composite - Mission Option (No Hohmann) - Launch 2024					
	Mass w/o Margin [kg]	Margin [%]	Margin [kg]	Total Mass [kg]	
Dry mass contributions					
Propulsion Module incl I/F PM-LM	**674.0**	**0.0**	0.0	**674.0**	
Lander	**600.9**	**20.0**	120.2	**721.1**	
Earth Return Vehicle	**220.0**	**20.0**	44.0	**264.0**	
Earth Re-entry Capsule	**35.0**	**0.0**	0.0	**35.0**	
Total Dry with Margin				**1694.1**	**kg**
Wet mass contributions					
PM Propellant	**1360.0**	**0.0**	0.0	**1360.0**	
LM Propellant	**415.0**	**0.0**	0.0	**415.0**	
ERV Propellant	**230.0**	**0.0**	0.0	**230.0**	
ERC Propellant	**N/A**	**N/A**	-	**-**	
Total Wet Mass				**3699.1**	**kg**
I/F Breeze - PM	**100.0**	**0.0**	0.0	**100.0**	
I/F PM - LM (incl in Dry Mass calculation)	**N/A**	**N/A**	-	**-**	
I/F LM - ERV (incl. in LM structure mass)	**N/A**	**N/A**	-	**-**	
Launch mass				**3799.1**	**kg**
Target Launch Mass				**5130.0**	**kg**
Below Mass Target by:				**1330.9**	**kg**

Table 8-16: Mass budget for baseline composite de-scoping Deimos mission (2024)

Phobos Sample Return Composite - Mission Option (No Hohmann) - Launch 2026					
	Mass w/o Margin [kg]	Margin [%]	Margin [kg]	Total Mass [kg]	
Dry mass contributions					
Propulsion Module incl I/F PM-LM	**674.0**	**0.0**	0.0	**674.0**	
Lander	**600.9**	**20.0**	120.2	**721.1**	
Earth Return Vehicle	**220.0**	**20.0**	44.0	**264.0**	
Earth Re-entry Capsule	**35.0**	**0.0**	0.0	**35.0**	
Total Dry with Margin				**1694.1**	**kg**
Wet mass contributions					
PM Propellant	**1505.0**	**0.0**	0.0	**1505.0**	
LM Propellant	**415.0**	**0.0**	0.0	**415.0**	
ERV Propellant	**230.0**	**0.0**	0.0	**230.0**	
ERC Propellant	**N/A**	**N/A**	-	**-**	
Total Wet Mass				**3844.1**	**kg**
I/F Breeze - PM	**100.0**	**0.0**	0.0	**100.0**	
I/F PM - LM (incl in Dry Mass calculation)	**N/A**	**N/A**	-	**-**	
I/F LM - ERV (incl. in LM structure mass)	**N/A**	**N/A**	-	**-**	
Launch mass				**3944.1**	**kg**
Target Launch Mass				**5302.0**	**kg**
Below Mass Target by:				**1357.9**	**kg**

Table 8-17: Mass budget for baseline composite de-scoping Deimos mission (2026)

8.6 System Options

The main architecture alternative evaluated during this assessment study is the choice of a **bipropellant propulsion system** for the landing module. This would lead to a series of advantages:

- Mass saving in the order of 250 kg due to the higher Isp of the bipropellant propulsion system (>290s vs 220s of hydrazine)
- Expected smaller propulsion subsystem even though more complex
- Slightly lower CoG at landing due to smaller tanks to be accommodated inside the LM.

Nevertheless this option would also present significant drawbacks:

- Phobos soil contamination problems would lead to the need of careful thruster accommodation, with possible thrust inefficiencies in case it is demonstrated that high tilt angles for thrusters are needed.
- Higher complexity of the propulsion subsystem would lead to higher costs (even though this must be traded off carefully against the possible need of a new development for hydrazine tanks for the baseline)
- Such a deep architecture change has to be implemented in early stages of the project to prevent heavy cost and schedule consequences.

It is therefore recommended to consider this option only in case the need to preserve the scientific observation of Deimos is evaluated as highest priority, and pending results of future studies concerning the mass at launch. This system option in fact could guarantee the feasibility of the mission with improved launch margins.

8.7 Backup Cooperation Scenario

A backup responsibility sharing between ESA and ROSCOSMOS has been assessed with the split already presented in Figure 8-1.

- ERC and PM under ESA responsibility
- ERV and LM under Russian responsibility

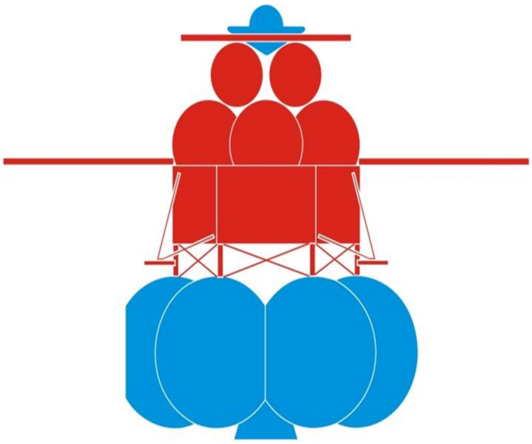


Figure 8-6: Backup responsibility sharing (ESA = blue, RUS = red)

The following assumptions have been considered when dimensioning the backup composite design:

- Same ERC as the baseline case
- Same ERV as the baseline case
- Russian LM extensively based on Phobos-Grunt re-use, with the following characteristics:
  - Dry mass without system margin: 838.4 kg
  - Maximum propellant capacity: 1060 kg
  - Propellant type: N_2H_4/NTO
  - Isp: 304 s
- European PM is based on the Phootprint design heritage (TAS-I design), with the following characteristics:
  - Dry mass: 469 kg (incl 20% system margin)
  - Propellant type: MMH/MON
  - Isp: 323 s
  - It has been checked the compatibility of the propellant tank size with the propellant budget for this backup scenario
- Adapter PM-LM: 60 kg
- Launcher adapter (Breeze-PM): 115 kg

The backup design is presented in the following figure:

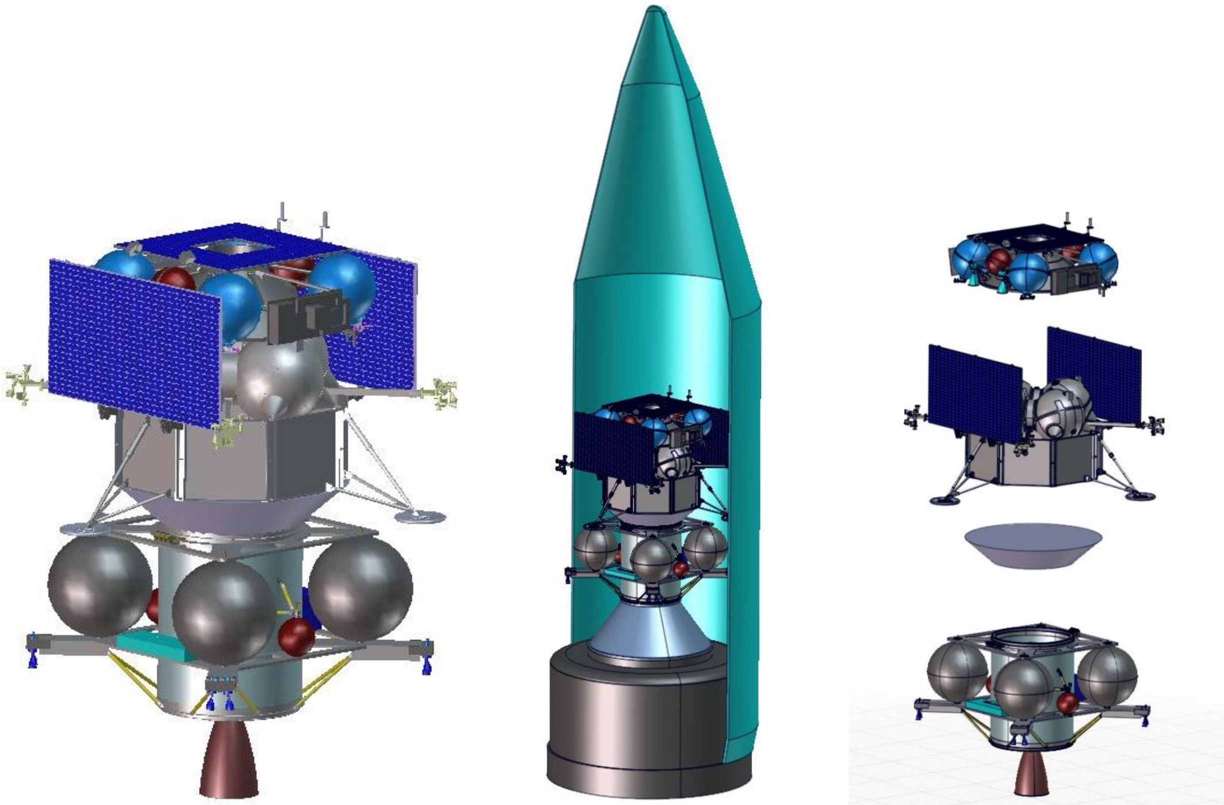


Figure 8-7 – View of the Phootprint backup cooperation scenario composite standalone (Left), under Proton fairing (Centre) and exploded (Right)

From the former drawings it is possible to appreciate some main differences between the Russian and the European concepts:

- The Russian ERV is based on a 3-footpads concept with a shorter octagonal structure and 4 main structural tanks. Only 2 single-panel solar wings are envisaged.
- The European PM, based on TAS-I Phootprint design, is based on a main central tube structure that supports a large central tank internally and four spherical tanks externally (as well as pressurant tanks and RCS thrusters)

In this case it is important to underline that the interfaces between modules under ESA responsibility are provisional (Launcher-PM and PM-LM). Simple conical structures have been assumed so far and conservative mass assumptions have been made, but a more detailed analysis is recommended in the upcoming phases of the project.

The mission timeline remains the same as for the baseline, but the high propellant capacity of the Russian LM, suggested to investigate a different staging with respect to the cooperation baseline, to take maximum advantage of the current design of the modules. In particular the chosen scenario in this case is the scenario 4C (i.e. jettisoning the PM at the arrival at Deimos, with the LM responsible for the Deimos-Phobos transfer)

	Low-Thrust manoeuvers	High Trust Engine Maneuvers			
		MOI	TOA-1 Deimos	TOA-2 Deimos	Deimos-Phobos
Scenario 4C	LM	PM	PM	PM	LM

Figure 8-8: Staging scenario chosen for the backup cooperation scenario

The delta-v considered for the outbound transfer is largely based on the same values of Table 8-9 with the addition of a provision for gravity losses (about 7%) on the manoeuvres performed by the PM. The Russian PM in fact, with its 20 kN engine, has a very high thrust-to-mass ratio that reduces gravity losses below 1%. A European development instead, could in the best case envisage the 1000 N HTAE (High Thrust Apogee Engine) therefore with this lower T/m ratio the loss contributions go up to approximately 7% for those manoeuvres that necessarily need to be performed in one single burn, such as the MOI. The TOAs indeed can be achieved through multiple small burns therefore the gravity loss effect can be minimised.

The overall baseline outbound delta-v up to Deimos, taking into account the corresponding gravity losses (GLs), on which the PM sizing is based, are then:

- **1734 m/s** for the 2026 launch scenario
- **1598 m/s** for the 2024 launch scenario

The global delta-v under the responsibility of the LM instead, is **1316 m/s**

The delta-v budget for the inbound case remains unchanged; therefore the overall ERC+ERV stack remains the same as the baseline.

The following tables show the overall mass budget for the backup cooperation scenario, computed with the following assumptions:

- Propellant masses have been computed taking into account 1.5 % residuals and 1% additional losses
- For the ESA PM a 20 % system margin on the dry mass has been applied
- For the Russian LM, as agreed with Lavochkin, given the very high heritage from Phobos-Grunt a 10% system margin on the dry mass has been applied.

It is clear how this scenario would lead to improved mass margins both in 2024 and 2026 launch scenario. On the other hand one needs to take these results with more care since this option was studied with a lesser level of detail. Of course also in this case the mass reduction option presented in section 8.5.4 (de-scoping of Deimos mission) is applicable, increasing dramatically the robustness of this backup scenario.

Phobos Sample Return Composite - Back-up Option 10% LM Margins Launch 2024					
	Mass w/o Margin [kg]	Margin [%]	Margin [kg]	Total Mass [kg]	
Dry mass contributions					
Propulsion Module	**391.0**	**20.0**	78.2	**469.2**	
I/F PM - LM	**60.0**	**0.0**	0.0	**60.0**	
Lander	**838.4**	**10.0**	83.8	**922.2**	
Earth Return Vehicle	**220.0**	**20.0**	44.0	**264.0**	
Earth Re-entry Capsule	**35.0**	**0.0**	0.0	**35.0**	
Total Dry with Margin				**1750.4**	**kg**
Wet mass contributions					
PM Propellant	**1848.0**	**0.0**	0.0	**1848.0**	
LM Propellant	**886.0**	**0.0**	0.0	**886.0**	
ERV Propellant	**230.0**	**0.0**	0.0	**230.0**	
ERC Propellant	**N/A**	**N/A**	-	**-**	
Total Wet Mass				**4714.4**	**kg**
I/F Breeze - PM	**115.0**	**0.0**	0.0	**115.0**	
I/F PM - LM (incl in Dry Mass calculation)	**N/A**	**N/A**	-	**-**	
I/F LM - ERV (incl. in LM structure mass)	**N/A**	**N/A**	-	**-**	
Launch mass				**4829.4**	**kg**
Target Launch Mass				**5130.0**	**kg**
Below Mass Target by:				**300.6**	**kg**

Table 8-18: Composite mass budget for backup cooperation scenario (Launch 2024)

Phobos Sample Return Composite - Back-up Option 10% LM Margins Launch 2026					
	Mass w/o Margin [kg]	Margin [%]	Margin [kg]	Total Mass [kg]	
Dry mass contributions					
Propulsion Module	**391.0**	**20.0**	78.2	**469.2**	
I/F PM - LM	**60.0**	**0.0**	0.0	**60.0**	
Lander	**838.4**	**10.0**	83.8	**922.2**	
Earth Return Vehicle	**220.0**	**20.0**	44.0	**264.0**	
Earth Re-entry Capsule	**35.0**	**0.0**	0.0	**35.0**	
Total Dry with Margin				**1750.4**	**kg**
Wet mass contributions					
PM Propellant	**2052.0**	**0.0**	0.0	**2052.0**	
LM Propellant	**891.0**	**0.0**	0.0	**891.0**	
ERV Propellant	**230.0**	**0.0**	0.0	**230.0**	
ERC Propellant	**N/A**	**N/A**	-	**-**	
Total Wet Mass				**4923.4**	**kg**
I/F Breeze - PM	**115.0**	**0.0**	0.0	**115.0**	
I/F PM - LM (incl in Dry Mass calculation)	**N/A**	**N/A**	-	**-**	
I/F LM - ERV (incl. in LM structure mass)	**N/A**	**N/A**	-	**-**	
Launch mass				**5038.4**	**kg**
Target Launch Mass				**5302.0**	**kg**
Below Mass Target by:				**263.6**	**kg**

Table 8-19: Composite mass budget for backup cooperation scenario (Launch 2026)

9 AEROTHERMODYNAMICS

9.1 Introduction

This chapter aims at reporting the preliminary aerothermodynamics assessment conducted within the Phobos Sample Return CDF study for the design of the Earth re-entry capsule.

The study was performed after two other CDF studies (MMSR and MMSR-A5) both assessing the feasibility and the preliminary design of a Phobos Sample Return mission. Many aerothermodynamics problems and solutions were carried over from the initial studies to the present one. Moreover, also industrial studies followed the initial CDF assessment.

Most of the material presented in the following pages has been directly extracted from the reports of the previous activities.

The work performed can be subdivided in three steps summarised hereafter in three different sections:

- Section 9.2 briefly explains the choice of the aeroshell shape from the initial trade-off to the selection of its final configuration (Section 9.2.1) concluding with a first aerodynamics characterisation of the chosen shape (Section 9.2.2)
- Section 9.3 contains the sensitivity analysis of the entry phase. Within this section 3 degree of freedom trajectory computations have been used to explore the variability of few key design quantities as a function of the ballistic coefficient and the Flight Path Angle (FPA). Section 9.3.1 will be dedicated to the assumptions, free parameters are listed in Section 9.3.2, constraints in Section 9.3.3 and finally the results are reported in Section 9.3.4
- Section 9.4 is dedicated to the baseline configuration selection where heat flux profiles have been extracted for TPS sizing.

9.2 Assumptions and Trade-Offs

9.2.1 Shape Trade-Off

The Phobos Sample Return capsule will perform a fully passive entry and descent (no supersonic nor subsonic parachute and consequently no shield separation) followed by a hard landing to minimise the design complexity and demonstrate the technology for a future Mars Sample Return capsule.

The high re-entry velocity of the mission has driven the design of the capsule to a blunt aero shell. Different probe shapes selected for previous (studied) missions are presented in Figure 9-1. They can be collected in different categories: sphere cones (with different semi-angle ranging from 45 to 70 deg), spheres and spherical sections.

As per MMSR, the shape trade-off has been made looking at the existing heritage, highlighting the commonalities and the main differences of already flown re-entry capsules with respect to the present study (see Figure 9-1 for a first visual comparison). When choosing the actual shape, the following points have been considered:

- To limit the complexity and reduce risk and costs, the capsule should be fully passive (meaning that supersonic and subsonic parachute should be avoided)
- The aeroshell is assumed to produce only drag and no lift
- The capsule shall provide enough volume to accommodate the sample container, the crushable foam and the necessary instrumentation
- The capsule shape shall guarantee stability in supersonic, transonic and subsonic regime
- The selected shape shall reduce the TPS mass fraction as much as possible
- A close similarity with MSR mission would be very beneficial for heritage and finally cost reasons.

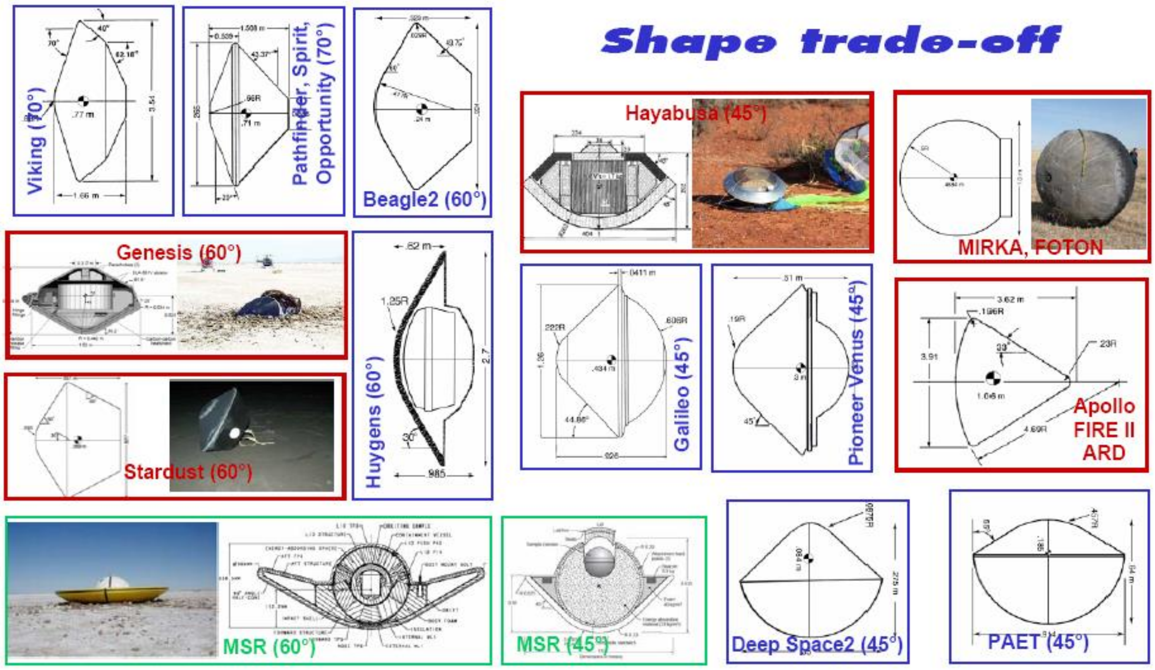


Figure 9-1: Shapes of (re-) entry capsule

A comparison of different shape/mission is summarised in Table 9-1. Here main discrepancies with the needs of the Phobos Sample Return have been indicated marking the cell in red, while commonalities are marked in green. It is important to pay attention to the following points:

- Spheres have been used in the past mostly for sub-orbital re-entry (a high penalty in TPS mass is expected when entering at higher velocities)
- Spherical sections (like Apollo and ARD) have been controlled, a feature which is not affordable within the Phobos Sample Return mission. An exception is FIRE II where the capsule was not meant to be (and was not) recovered
- Sphere cones with a half angle of 60 deg (like Stardust and Genesis) made use of supersonic parachutes (due to transonic instability) which should be avoided for the mission under consideration.

It becomes clear from Table 9-1 that, among the considered shapes, the only re-entry capsule shape which has no major discrepancies with the needs of the Phobos sample Return mission is the one of Hayabusa.

Consequently, it has been decided to select the shape of Hayabusa as reference for a deeper analysis of the mission knowing that a complete characterisation of the capsule performance, especially the stability in transonic, will have to be undertaken in the next phase.

	Previous mission	Base diameter (m)	Nose radius (m)	Mass (kg)	Entry velocity (km/s)	FPA (deg)	Max heat flux MW/m2	Super para	Sub para	Remarks
Sphere-cone 45 deg	Hayabusa	0.404	0.202	16.27	11.3	-13.8	15	-	Cross	
Sphere-cone 60 deg	Genesis	1.51	0.43	210	10.8	-8	7	DGB	Parafoil	
	Stardust	0.827	0.23	45.8	12.6	-8.2	12	DGB	Triconical	
Sphere	Photon	2.3	1.15	2472	7.6	-2	1.9			
	Mirka	1	1	154	7.6	-2.5	1.2			
Spherical section	Apollo (4)	3.9	4.69	5424	10.73	-7.1	4.9	-	Conical, ribbon, ringsailes	controlled
	Fire II (Initial conf.)	0.67	0.935	86.5	11.35	-14.7	11.4	-	-	
	ARD	2.8	3.36	2715	7.54	-2.6	1.2	-	Flat-Ribbon Conical-Ribbon Slotted-Ribbon	controlled

Table 9-1: Comparing different missions: red indicates a main discrepancy with the need of Phobos Sample Return mission, green a close similarity

9.2.2 Initial Aerodynamic Database for the Selected Shape

The aerodynamic database available in RD[5], which has been then adopted in the study, is reported hereafter in Figure 9-2.

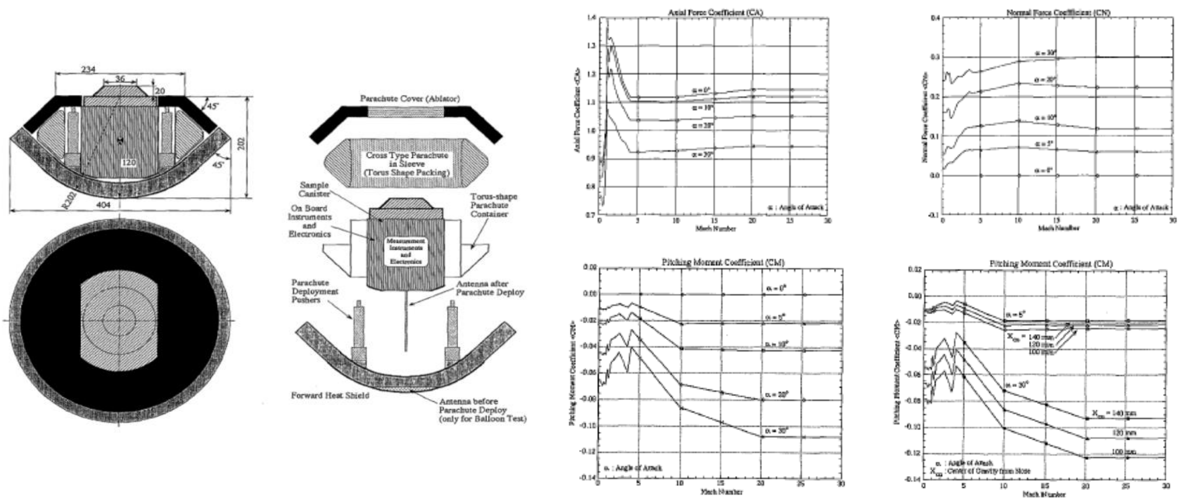


Figure 9-2: Hayabusa geometry and adopted aerodynamic database as in RD[5]

9.3 Sensitivity Analysis

9.3.1 Assumption

The following assumptions have been adopted in the parametric study:

(A1) The aerodynamics coefficients of the entry probe have been assumed to be equal to the ones of Hayabusa presented above.

(A2) The front body shape of Hayabusa (with 45° half angle blunted cone) with a (initial) nose diameter of 0.410 m has been selected.

(A3) In support to the aerothermodynamics calculations, the 3 degree of freedom TRAJ3D code (RD[6]) has been used. Inputs for the trajectory code are: the atmospheric profile (temperature, density and pressure as function of altitude), characteristic of the probe and potentially of parachutes (mass, diameter, drag coefficient…) and the entry conditions (velocity vector at interface altitude).

(A4) Different profiles of the earth atmosphere are available in the literature as the GRAM model (RD[7]) and US 1976 standard model (RD[8]): to simplify comparison with existing data, the last one has been used in the computations.

(A5) The Detra-Hidalgo (RD[9], valid for velocities below 9 km/s) and Tauber-Sutton (RD[10], valid from 9 to 16 km/s) formulation has been used to estimate the radiative heat flux, while Detra and Hidalgo (RD[9]) have been adopted to calculate the convective contribution. Below the heat fluxes are given in W/cm², V indicates the upstream velocity (m/s), $Nose_D$ is the nose diameter (m) and ρ is the upstream density expressed in kg/m³

- Detra and Hidalgo (Radiative heat flux for V < 9 km/s):

$$q_{rad}^{DH} = 113.5 \cdot \left(\frac{Nose_D}{0.6096} \right) \cdot \left(\frac{\rho}{1.22522} \right)^{1.6} \cdot \left(\frac{V}{3.048 \cdot 10^3} \right)^{8.5}$$

- Tauber and Sutton (Radiative heat flux for 9 < V < 16 km/s):

$$q_{rad}^{TS} = 4.736e4 \cdot \left(\frac{Nose_D}{2} \right)^{AEXP} \cdot \rho^{1.22} \cdot F(V)$$

with *F(V)* tabulated function and AEXP function of the velocity and the density and the $Nose_D$ (see e.g. RD[10], RD[11]).

- Detra and Hidalgo (Convective heat flux):

$$q_{conv}^{DH} = 1.135 \cdot 8.65e2 \cdot \left(\frac{Nose_D}{0.6096} \right)^{0.5} \cdot \left(\frac{\rho}{1.22522} \right)^{-0.5} \cdot \left(\frac{V}{3.048 \cdot 10^3} \right)^{3.15}$$

9.3.2 Parameters

Within the entry phase, special attention has been paid to investigate the influence of the entry conditions and the design configuration on the (convective and radiative) heat fluxes (and heat loads), maximum deceleration and stagnation pressure experienced by the probe. To this end, the following quantities have been left free in the parametric analysis:

(B1) Flight path angles (FPA) ranging from -5 deg to -20 deg have been considered in the analysis (but in the following section the results are presented for FPA > -14 deg: according to the simulation findings, the heat fluxes for smaller FPAs are well above the TPS capability).

(B2) Ballistic coefficient. Different design configurations of the entry capsule have been included into the parametric analysis considering different ballistic coefficients[1] ranging from 40 kg/m² to 150 kg/m² (results hereafter have been reported only for ballistic coefficients between 40 and 100 kg/m² covering the area of interest).

9.3.3 Constraints and Design Driver

The main requirements/constraints of the Phobos Sample Return entry probe, which have been applied to the parametric analysis, are briefly summarised hereafter:

(C1) In line with the worst case scenario foreseen by mission analysis, the relative entry velocity has been fixed to 12.3 km/s at the interface altitude of 120 km.

(C2) The ablative material under development shall be able to withstand peak heat flux levels up to 14-15 MW/m². Consequently within the parametric analysis the (total) maximal heat fluxes shall be confined below 15 MW/ m² (although further development and verifications could increase the material capability to around 18 MW/m²).

(C3) Stagnation (and dynamic) pressure shall be restricted by material and structure capabilities. The TPS under development (see Chapter 20 Thermal) shall be specified for up to 800 mbar (80000 N/m²) stagnation pressure at

[1] The (hypersonic) ballistic coefficient B_c is defined by $B_c = M/(C_d * A_{Ref})$ where M is the mass, C_d is the (hypersonic) drag coefficient and A_{Ref} is the base area of the entry capsule.

maximum heat flux and consequently the stagnation pressure experienced by the capsule shall be confined below 800 mbar at maximum heating.

(C4) Maximum deceleration shall be restricted for structure limitation below 80 g (eventually increased to 100 g).

(C5) The landing velocity should be such that the appropriate choice of a crushable material would limit the maximum deceleration of the sample container below 2000 g (possible as low as 800 g).

9.3.4 Parametric Results

The results of the different trajectories defined by (B1)-(B2) and (C1) are summarised in Figure 9-3 to Figure 9-7 where the corresponding quantities are plotted as a function of the ballistic coefficient (horizontal axis) and FPA (vertical axis).

In Figure 9-3, the results of the skip-out analysis are reported: the blue area Earth Return Capsule (ERC)indicates skip-out, the green implies that the FPA crosses the -2 deg while the is descending (close to skip-out or skip entry). Finally, for any combination of FPA and ballistic coefficient in the red area the Phobos Sample Return capsule enters "normally" into the earth atmosphere.

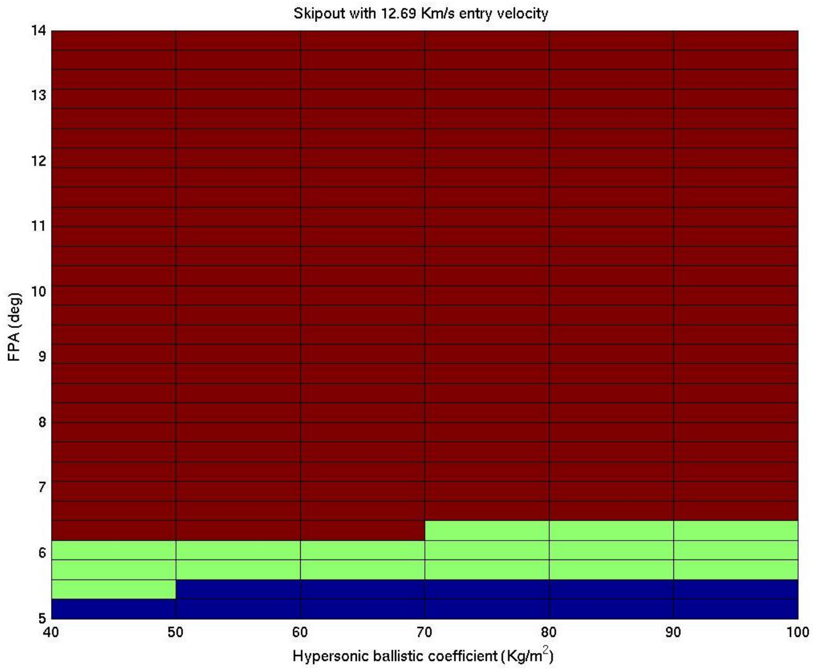


Figure 9-3: Skip-out

In Figure 9-4 the maximum total (convective plus radiative heat flux) at stagnation point is plotted: margins (20% on convective contribution and 100% on radiative one) are included.

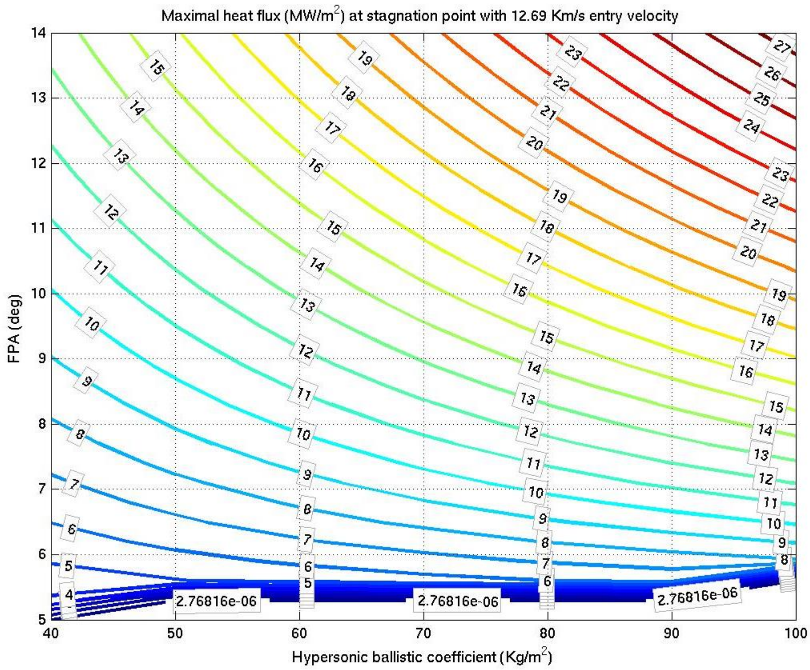


Figure 9-4: Heat (convective and radiative) flux at stagnation point

Integrated heat load (including margin) is plotted in Figure 9-5, while Figure 9-6 reports the stagnation pressure at maximum heat flux.

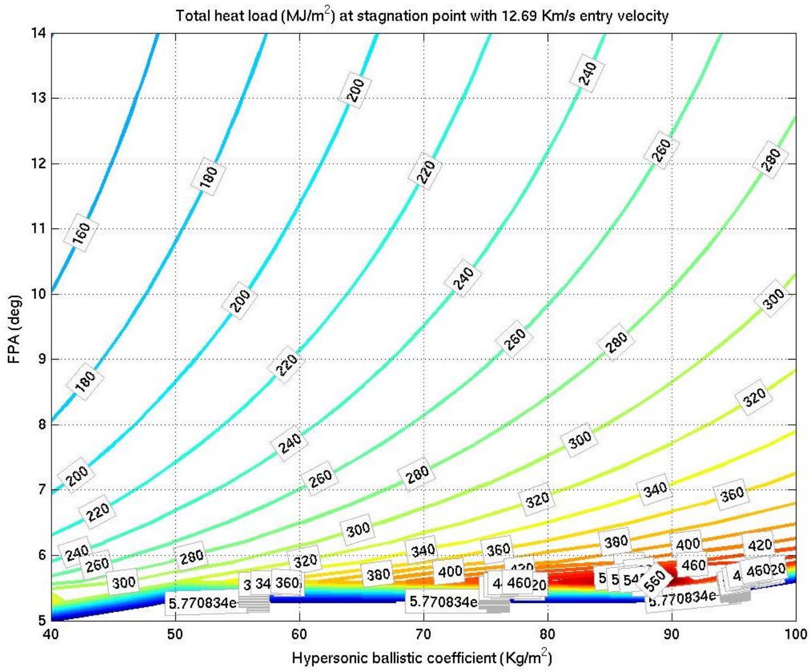


Figure 9-5: Integrated heat load (at stagnation point)

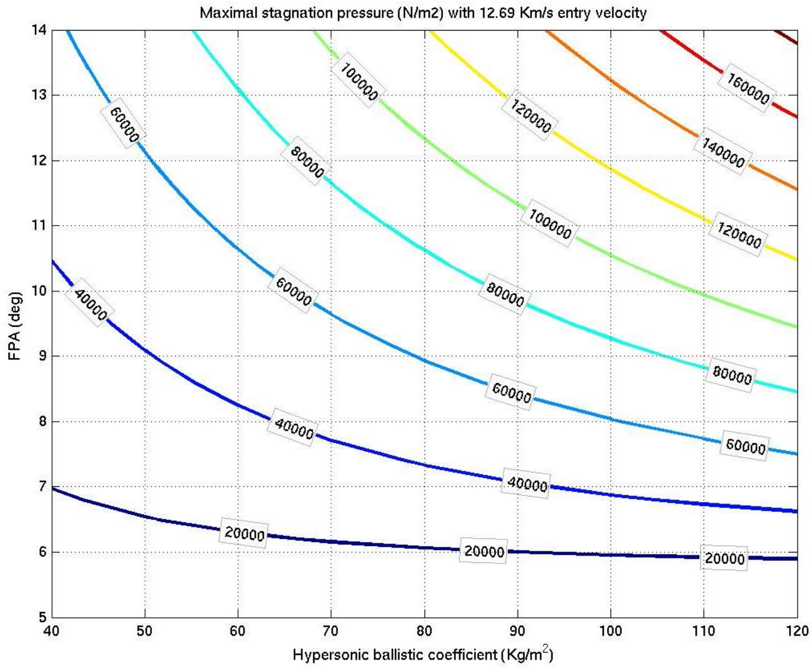


Figure 9-6: Stagnation pressure (at maximum heat flux)

On Figure 9-7 the maximum deceleration experienced by the ERC is summarised: very light dependency on the ballistic coefficient can be noted while the major role is left to the FPA (see e.g. RD[12]).

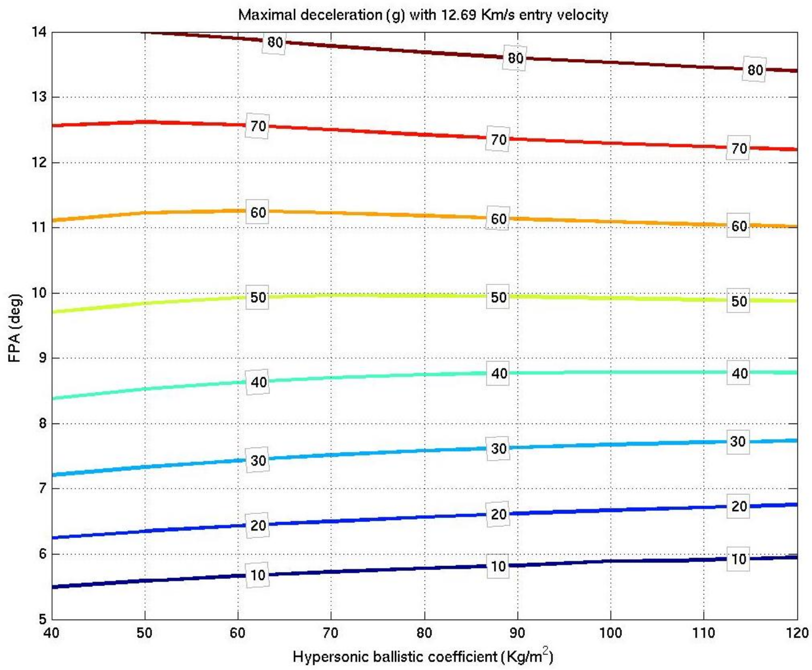


Figure 9-7: Maximum deceleration

Finally, the results of the parametric analysis have been used to select a feasible domain (the possible ballistic coefficient / FPA combinations that fulfil the requirements) illustrated in Figure 9-8 (bottom centre picture indicated by the white area): the constraints (C2) - (C4) are superimposed on the results presented above and a maximum FPA of -8 deg has been indicated. No restrictions have been applied on the total heat load but this should be limited by mass constraints.

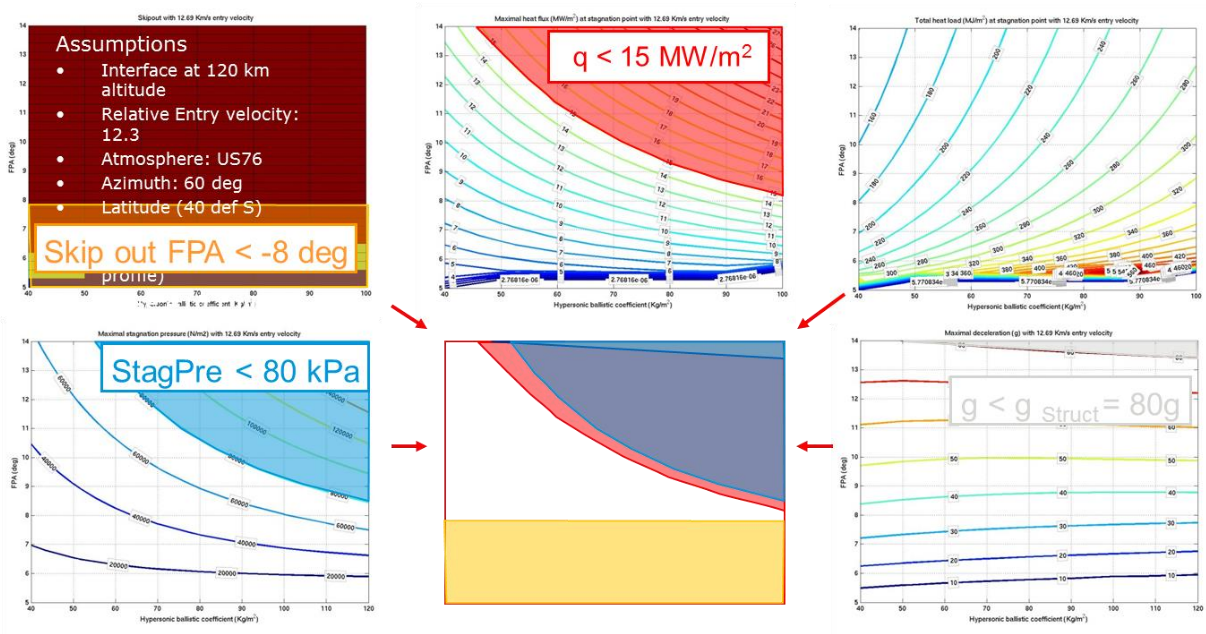


Figure 9-8: Feasible domain definition (bottom centre picture indicated by the white area)

9.4 Baseline Design

In this section a brief summary of the ERC design is given and the associated re-entry trajectory profiles for TPS sizing are presented.

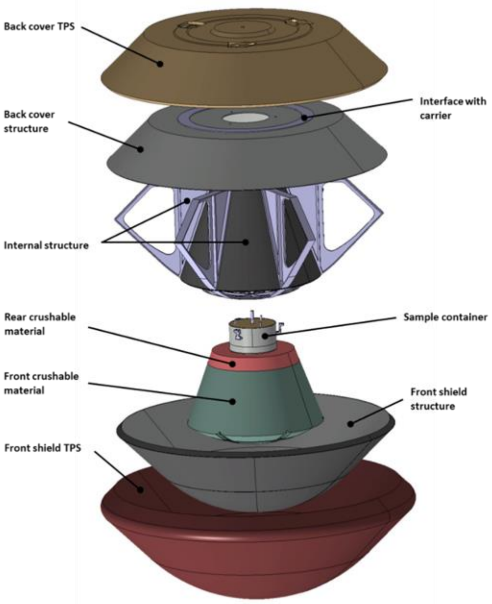
While in the parametric analysis the size of the ERC has been implicitly considered within the ballistic coefficient and the nose diameter being fixed (at 410 mm), in the present sections a short overview of the ERC design is provided.

In Section 9.2.2, it has been decided to select the shape of Hayabusa as reference. Consequently only a scaling is applied here to define the actual dimension of the Phobos Sample Return capsule.

With the support of the results presented above and taking into account system requirements, a baseline ERC configuration has been selected with the following design properties:

- Diameter = 750 mm
- Height = 375 mm (height / diameter = 0.5)
- Mass = 26.3 kg (31.5 kg with DMM)
- XCoG = 209.8 mm nominal

- XCoG / Diameter = 27.97% (if stability requires, improvements can be obtained with ballast and/or back cover modifications)



Parameters	Preliminary Design
Aeroshape	HAYABUSA
Diameter D	750 mm
Mass	≈ 35 kg (with DMM & SM)
Centring	< 30%D
Max g-load	≈ 1300g

Figure 9-9: ERC preliminary design (ref: Airbus DS Phootprint design)

9.4.1 Baseline Trajectory for TPS Sizing

For TPS sizing purposes, beside the nominal one, also a maximum heat flux and a maximum heat load trajectories have being computed and reported in Figure 9-10.

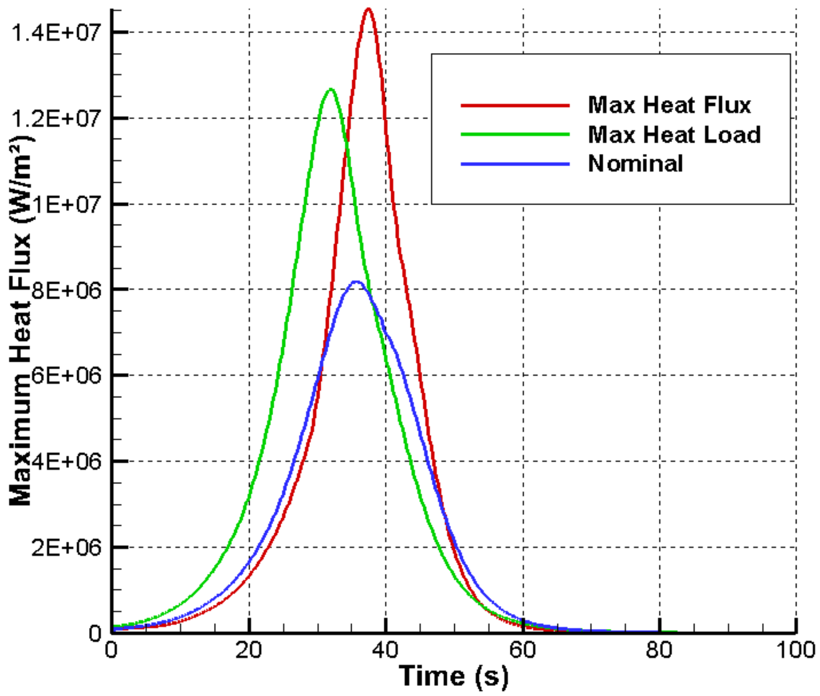


Figure 9-10: nominal, max heat flux and max heat load trajectories (incl. margins)

Max heat flux trajectory has been considered for TPS Ablator selection, max heat load trajectory for TPS thickness computation. Table 9-2 summarises the salient characteristics of the three derived trajectories.

Trajectory	Max. Heat Flux	Nominal	Max. Heat Load
FPA	Maximum (Steep)	Nominal	Minimum (Shallow)
FPA (deg)	-10	-9.8	-9.6
Maximum Heat Flux (MW/m²)	15	8.2	12.7
Maximum Heat Load (MJ/m²)	221.3	178.8	235.8
Total Pressure at peak heat flux (kPa)	36.4	25.3	30.1
Maximum deceleration during entry (g)	60.5	42.2	44.0
Impact velocity (m/s)	41	38	37

Table 9-2: Trajectory principal characteristics

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10 CONFIGURATION

The Phobos Sample Return mission consists of four elements: the propulsion module (PM), the landing module (LM), the Earth return vehicle (ERV), and the Earth re-entry capsule (ERC) as shown in Figure 10-1.

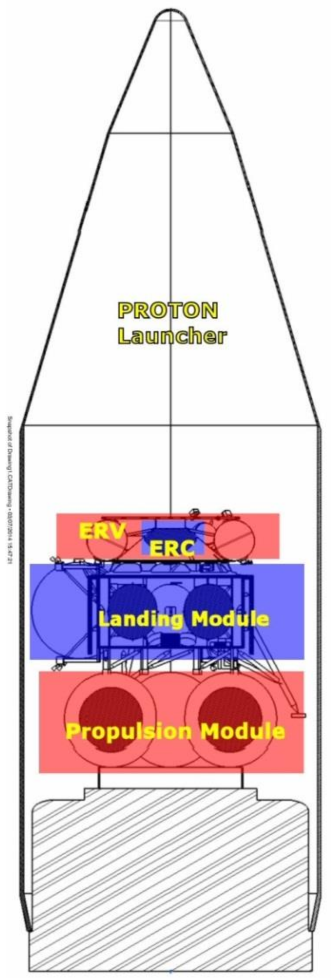


Figure 10-1: Phobos-SR elements

Main task of the configuration subsystem is to design a new LM for this mission. The design was generated based on the following requirements and design drivers.

10.1 Requirements and Design Drivers

- Phobos Sample Return stack has to fit inside the available volume of the Proton fairing
- LM will interface with the propulsion module by means of 8 available interface points of the Fregat upper stage

- ERV will be placed on top of LM by means of a conical interface (under Russian lead) with dimension 2m/1.6m/.3m (bottom diameter, top diameter and height of the cone)
- LM shall accommodate subsystems equipment based on their requirement.

10.2 Baseline Design

Phobos Sample Return LM uses a simple beam element adapter as shown in Figure 10-2 between the PM-LM. The octagonal shape of the LM is derived from this 8 I/F point adapter with Ø2m diameter. This is done to have a simple load path from the LM to the PM.

Figure 10-2: Adapter PM-LM integrated on Fregat upper stage

Figure 10-3 shows the stowed configuration of the LM. The outside diameter of 2.3m of the octagonal shape box is used to provide support to all subsystem equipment.

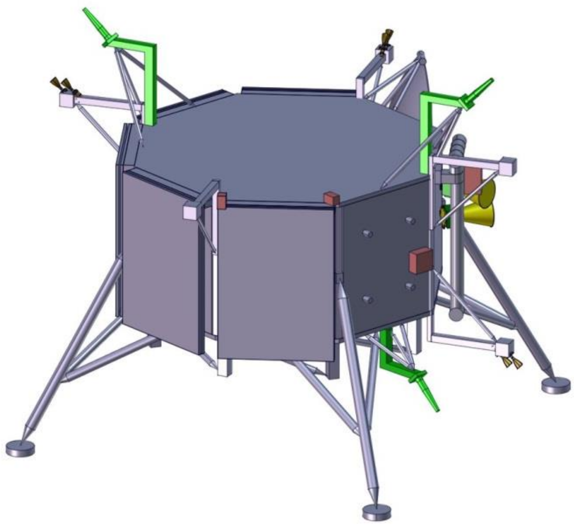


Figure 10-3: Phobos-SR – LM stowed configuration

The LM accommodates the subsystem equipment as follows:

Propulsion S/S:

There are 4 propulsion tanks with 0.7m diameter accommodated inside the LM. The tanks require about 1m height for the purpose of accommodating the pipe lines. This requirement is sizing the height of the octagonal shaped LM. There is no pressurant tank needed to be accommodated.

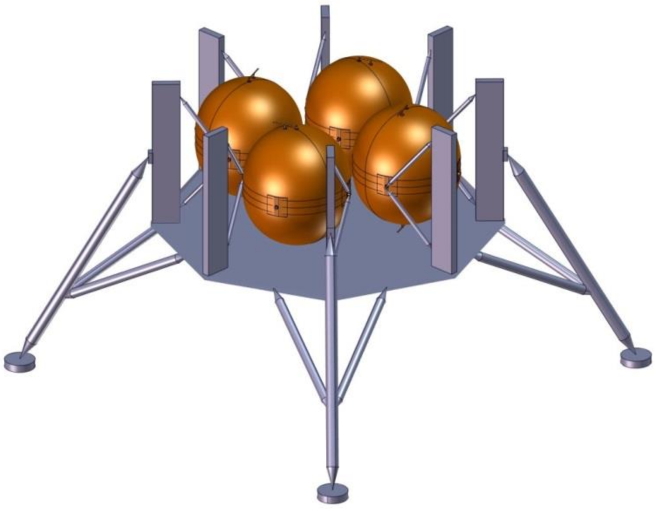


Figure 10-4: LM Propulsion tanks accommodation

Power S/S

The Power subsystem requires about 10m² solar panel areas. The side panel of the LM has a dimension of 825mm by 1m. To provide the required surface area, each solar panel will have longer dimension than the LM side panel namely 1.23m. This will then give 3 free surface areas that can be used to provide mounting area for other equipment. The electronic equipment of the power S/S, one unit of PCDU and one unit of battery are placed inside the LM.

Communication S/S

A High Gain Antenna (HGA) of 1.4 m diameter is accommodated on the LM side panel with 2 DoF mechanisms support. Several LGA's are needed to cover a hemispherical field of view. During in-orbit operation the LM cannot have fully hemispherical FoV because the PM is still attached under the LM. In this case the LM could have the possibility to use on-board LGA of the PM. The same panel that supports the HGA accommodates also other COMMS equipment boxes. The COMMS boxes are illustrated in green in Figure 10-5.

GNC S/S

There are in total 16 x 20N thrusters required for the GNC S/S. Thrusters need to be located outside the spacecraft body as far as possible to have enough moment for GNC to control the spacecraft. The GNC items are illustrated in pink in Figure 10-5.

Data Handling S/S

The onboard computer (light blue in Figure 10-5) is internally mounted on one of the side panels.

Robotics

Robotic arm with stowage dimension of 1300 mm x 410 mm x 320mm is accommodated on a free side panel as shown in Figure 10-5. There are 3 configurations of the robotic arm shown in the picture namely: stowed (grey), sampling (red) and sample transfer to ERC configuration (yellow).

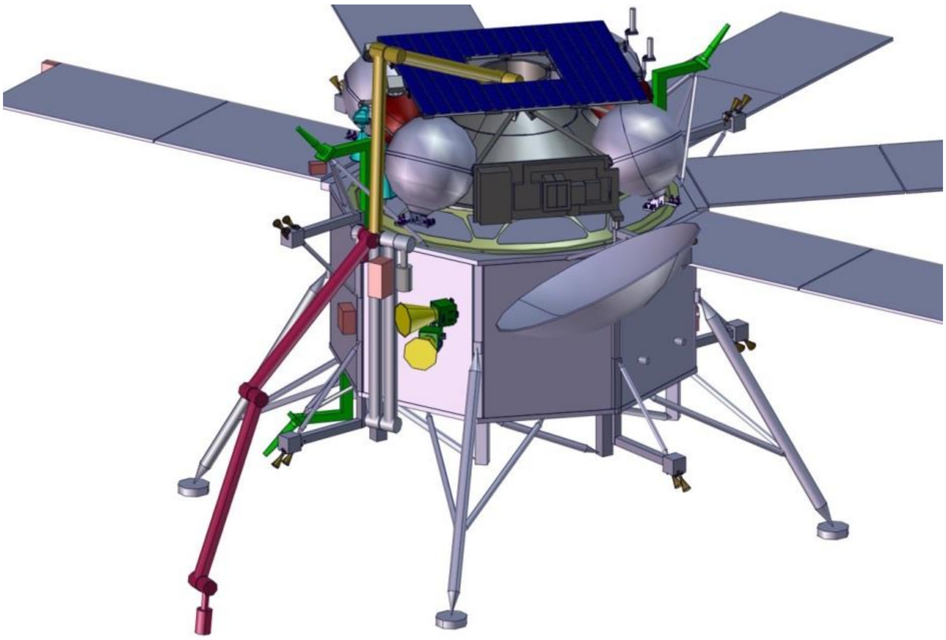


Figure 10-5: Phobos-SR – LM – 3 configurations of the Robotic arm

Payload

Payload instruments are placed inside the spacecraft. One dedicated instrument panel provides opening windows for each of the instruments to meet their pointing requirement as shown in Figure 10-6.

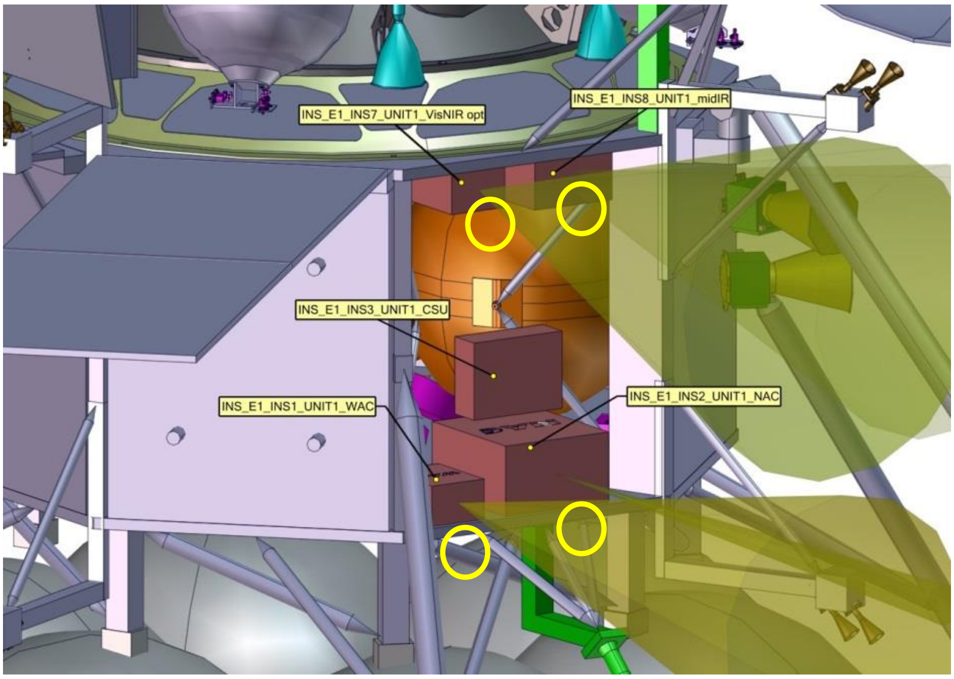


Figure 10-6: Phobos-SR – LM – PL Field of view

Figure 10-7 and Figure 10-8 show the accommodation for Phobos Sample Return LM and its internal and external accommodation respectively.

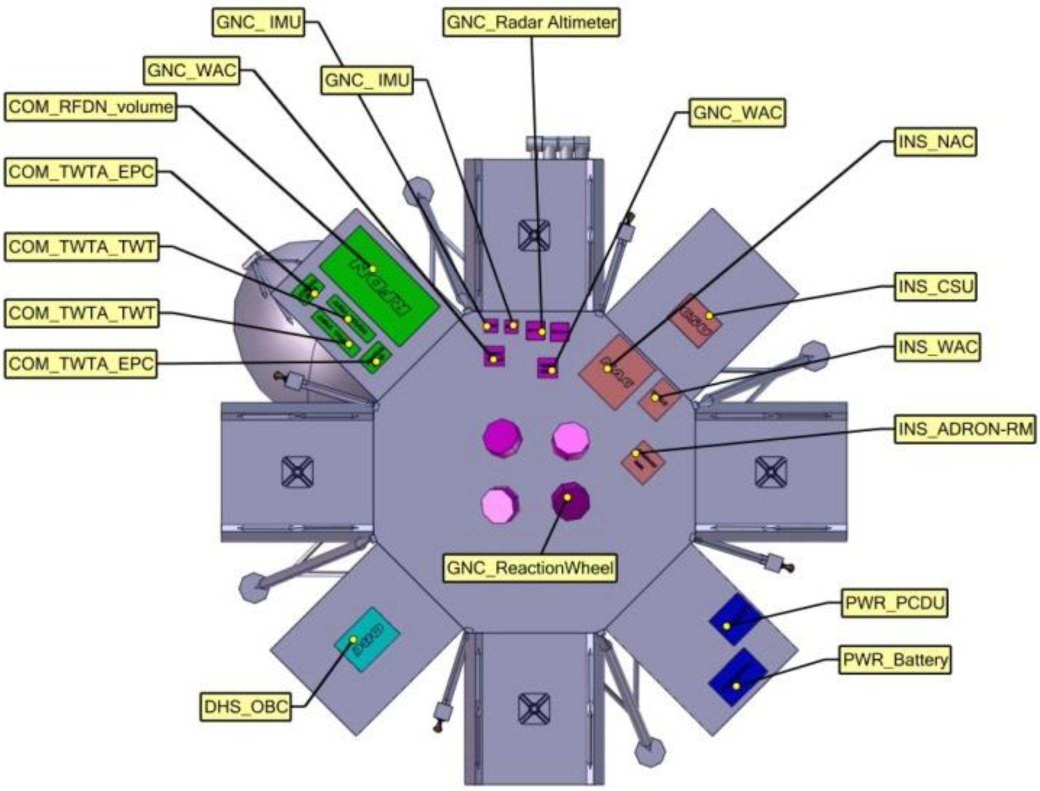


Figure 10-7: Phobos-SR – LM internal accommodation

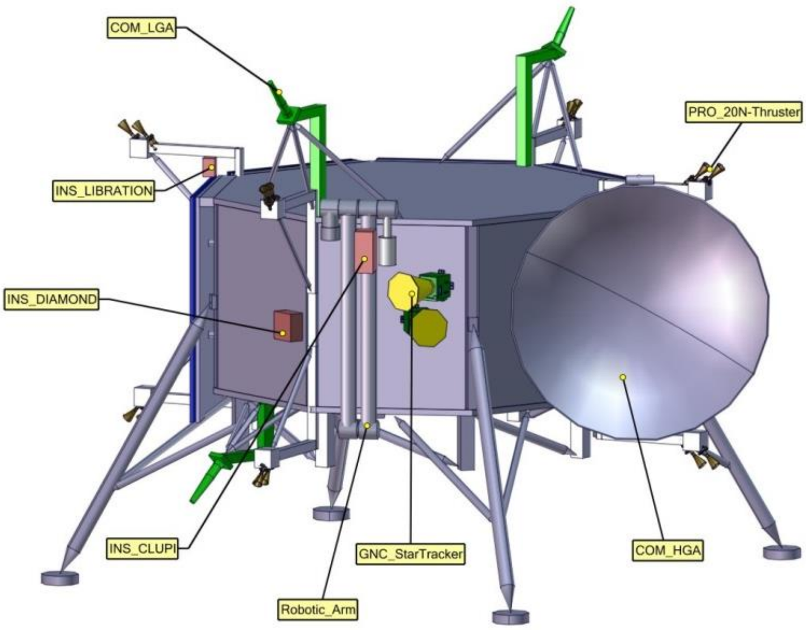


Figure 10-8: Phobos-SR – LM external accommodation

10.3 Overall Dimensions

The following figures show the overall dimension of the Phobos Sample Return LM for stowed configuration with maximum dimension of 4.0 mm x 2.8 m x 4.3m

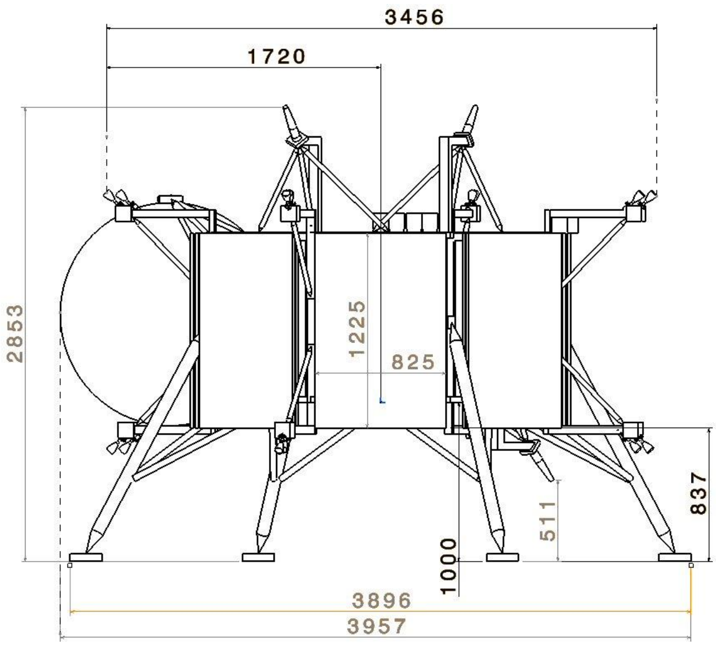


Figure 10-9: Phobos-SR – LM dimension in stowed configuration – side view

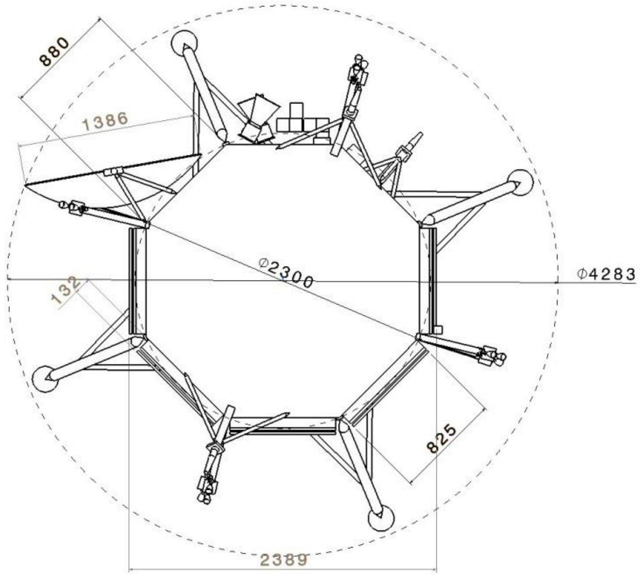


Figure 10-10: Phobos-SR – LM dimension in stowed configuration – top view

For deployed configuration, dimension can be read from Figure 10-11 and Figure 10-12.

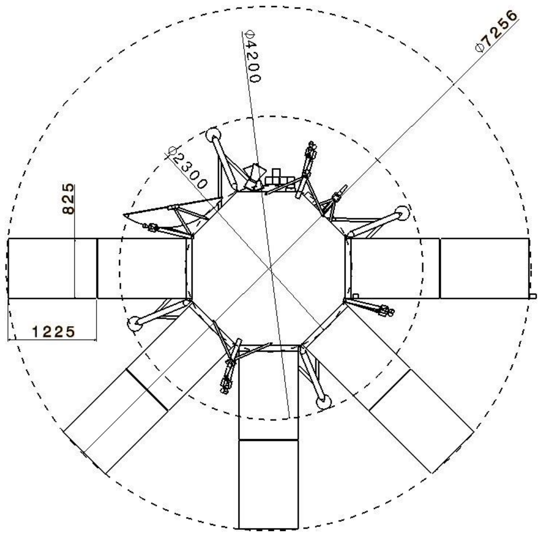


Figure 10-11: Phobos-SR – LM – dimension in deployed configuration - top view

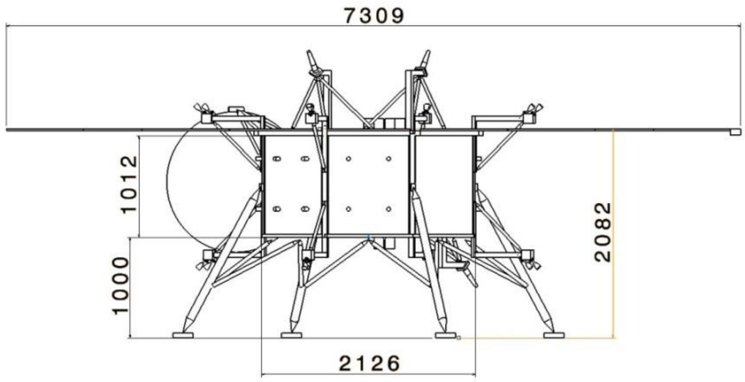


Figure 10-12: Phobos-SR – LM – dimension in deployed configuration - side view

11 STRUCTURES

11.1 Requirements and Design Drivers

The following requirements and design drivers were used during this study to build the MMSR composites:

SubSystem requirements		
Req. ID	**STATEMENT**	**Parent ID**
STR-010	The structure of the different modules shall provide support for all their elements	
STR-020	The structure shall withstand all loads during lifetime	
STR-030	The structural stiffness shall guarantee minimum lateral and longitudinal fundamental frequencies compatible with the launcher requirements	
STR-040	The structural mass shall be as low as possible	

11.2 Assumptions and Trade-Offs

Following the study baseline, Propulsion Module and Earth Return Vehicle are provided by ROSCOSMOS, and Landing Module and Earth Return Capsule are provided by ESA.

As such only the Landing Module (LM) and the adapters to the Propulsion Module (PM) below, and to the Earth Return Vehicle above are discussed here.

11.2.1 LM Structure

Different options have been investigated for the layout and the materials of the Landing Module structure.

Two possible designs were identified:

Option A A cylindrical central, primary load carrying structure (with tanks and bottom/top/lateral panels outside)

Option B An octagonal-shaped primary load carrying structure (with tanks and bottom/top panels inside)

The following rationale has been followed:

- The mass of the LM shall be as low as possible
- PM upper interface diameter is 2 m, 8 points
- ERV lower interface diameter around 1.62 m in order to allow using it with a rather large diameter ERC
- The number and dimensions for the tanks comes from propulsion requirements.

Option A

- The diameter change from 2 m of the PM upper I/F to a 1.194 m diameter of the LM lower I/F, including conversion of a 8 point load to a distributed flux with an overflux in the order of 20-30% maximum is being taken care of by an 8-point/conical structure as shown in Figure 11-1 (the upper part is derived from a PAS1194 adapter, the lower part would have to be designed specifically for the Phobos mission).

Figure 11-1: Option A, adapter PM-LM

- The cylindrical central tube would be rather simple and light, considering that the flux would be quasi-uniform along the circumference
- At the height of lower and upper tube interface a bottom and top sandwich panel would be foreseen, both of hexagonal or octagonal shape
- Bottom and top panels would be connected at the outer edges by 6-8 lateral CFRP/aluminium honeycomb sandwich panels and/or stiffened aluminium panels to accommodate radiators
- The tanks would be located between tube and lateral panels
- On top of the tube, an inverse conical adapter would form the adapter from LM to ERV.

Figure 11-2 shows a sketch of the configuration and gives rough mass values (note: these mass values originate from the study beginning and have been updated in the course of the study).

The total mass for LM, PM-LM adapter and LM-ERV adapter is around 212 kg. The main contributors are the LM, which is higher than in option B (see below), and the PM-LM adapter (mass estimated based on PAS 1194VS) which has to provide a “small over-flux” upper interface.

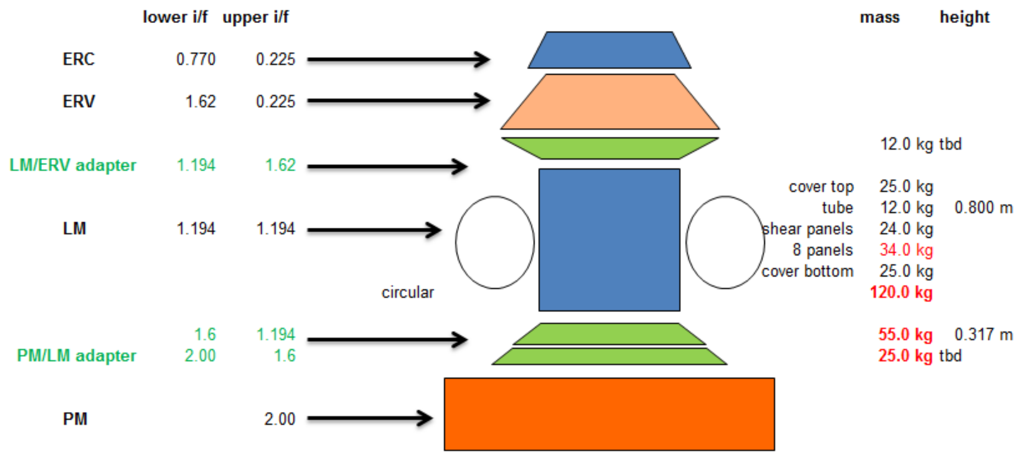


Figure 11-2: Option A, configuration sketch and mass estimates

Option B

- No change in diameter, i.e. maintaining 2 m / 8 points configuration, for PM and LM; no distributed flux with limited overflux for LM required
- The primary load path is of octagonal shape and consists of eight lateral panels plus eight struts/connectors
- Top and bottom are closed by panels
- Four tanks are located inside the octagonal
- PM-LM adapter as proposed by ROSCOSMOS, see Figure 10-2
- LM-ERV adapter as shown in Figure 11-8 transferring the loads from the 8-point load path to a distributed flux with an overflux in the order of 20-30% or less.

Figure 11-3 shows a sketch of the configuration and gives rough mass values (note: these mass values originate from the study beginning and have been updated in the course of the study).

The total mass for LM, PM-LM adapter and LM-ERV adapter is around 138.5 kg.

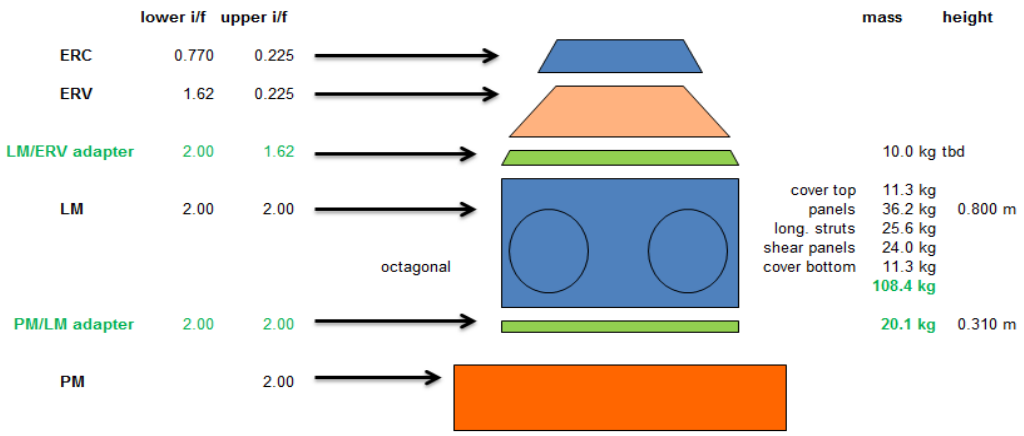


Figure 11-3: Option B, configuration sketch and mass estimates

Option B is interesting if the propulsion tanks fit inside, and also if the upper interface to the ERV has a rather large diameter, larger than the diameter of the central cylinder; both conditions are met.

Option B is selected because

- The four propellant tanks fit inside a 2 m diameter octagonal prism
- The 8 interface points from Fregat make an octagonal shape an obvious choice
- The design provides a direct load path from ERV to PM via the LM
- The mass is lower than for *option A*, mainly due to a simpler load path
- The design provides higher stiffness
- The eight point interface of Fregat, on which the PM is based, continues as load path through the LM; peak fluxes can be easily taken care of by dedicated design of the connectors of the 8 lateral panels.
- The eight point load path of the LM can be "smoothed" to a distributed flux by the LM-ERV adapter which is smaller and has lower strength requirements than the PM-LM adapter; hence offers an extra mass saving.

11.2.2 Adapter PM-LM

Two adapter designs have been considered, the rationale is presented in the previous section 11.2.1 together with the LM structure.

The design chosen is shown in Figure 10-2.

11.2.3 Adapter LM-ERV

An inverted cone based on a typical Ariane5 or Vega adapter would be required for *Option A* of section 11.2.1. Due to the different strength requirements, a custom-design would be required.

The baseline adapter (for Option B) is described below.

11.3 Baseline Design

An overall view of the Phobos Sample Return baseline design is shown below (two lateral panels and bottom panels are removed for visibility).

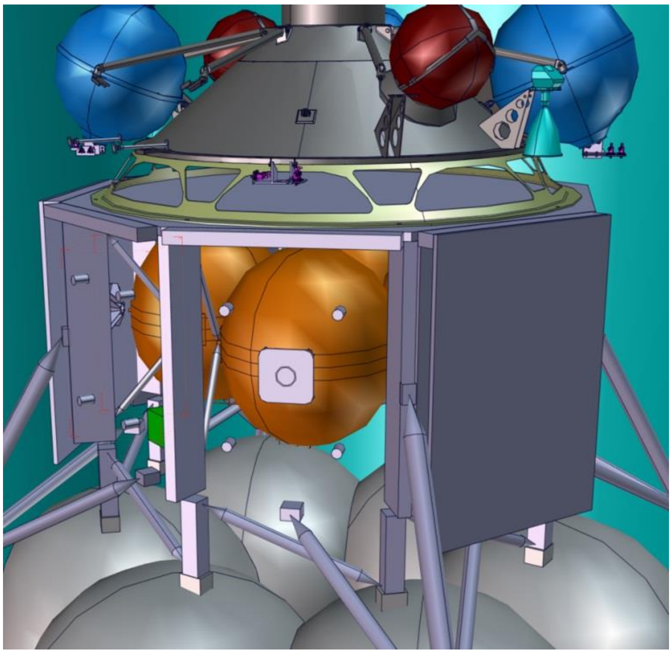


Figure 11-4: LM, adapters and ERV baseline

11.3.1 LM Structure

Based on the trade-off in section 11.2.1, the following design has been chosen.

- 2 m diameter octagonal shaped geometry allows minimum mass of the primary structure
- Lateral panels contribute to the primary load path together with longitudinal struts, which also serve as lateral panel connectors
- Bottom and top covers provide shear stiffness to the primary structure and attachment opportunities for equipment
- An 8-point/conical LM-ERV adapter allows fitting of different interface diameters and flux reduction from 8 point-overflux to near-constant flux
- The overflux from the PM is transferred/distributed to the ERV at the upper interface via a rather small custom-designed adapter
- Tank supports are polar mounted with struts & fittings.

Figure 11-4 and Figure 11-5 show the main structural elements of the LM:

- Two lateral panels (w/ radiators) made of aluminium face sheets /aluminium honeycomb (or alternatively aluminium panels with stiffeners)
- Six lateral panels (w/o radiators) made from CFRP phase sheets / aluminium honeycomb
- One top panel (CFRP phase sheets / aluminium honeycomb) & aluminium ring of octagonal shape
- One bottom panel (CFRP phase sheets / aluminium honeycomb) & aluminium ring of octagonal shape

- Eight struts/connectors for the lateral panels in aluminium
- Four struts per tank plus one fitting in aluminium.

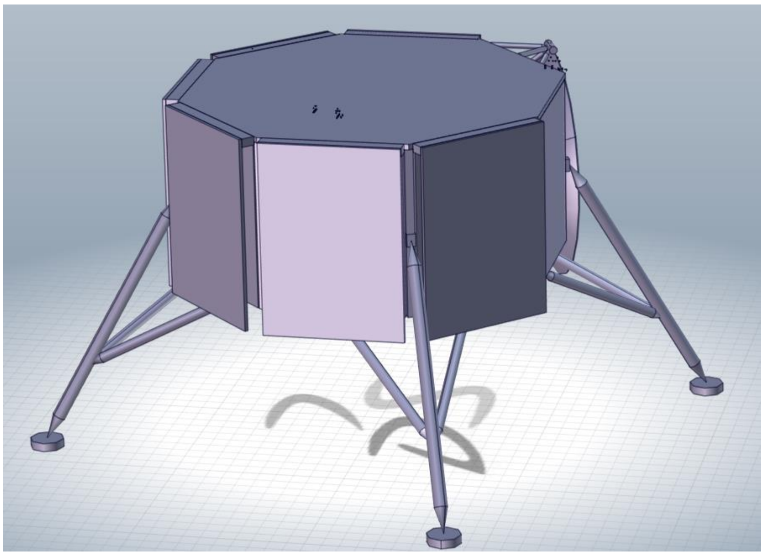


Figure 11-5: LM main structural elements and legs

11.3.1.1 LM strut design

Each of the four propellant tanks is supported by two bipods attached to the equator on each side, plus a fitting to the lateral panel providing the missing rotational stiffness.

The structural dimensioning is shown in Figure 11-6. The QSL used is 4.2 g coming from the Proton User Guide RD[13]. Adding a safety/uncertainty factor, a QSL of 10 g is used for dimensioning.

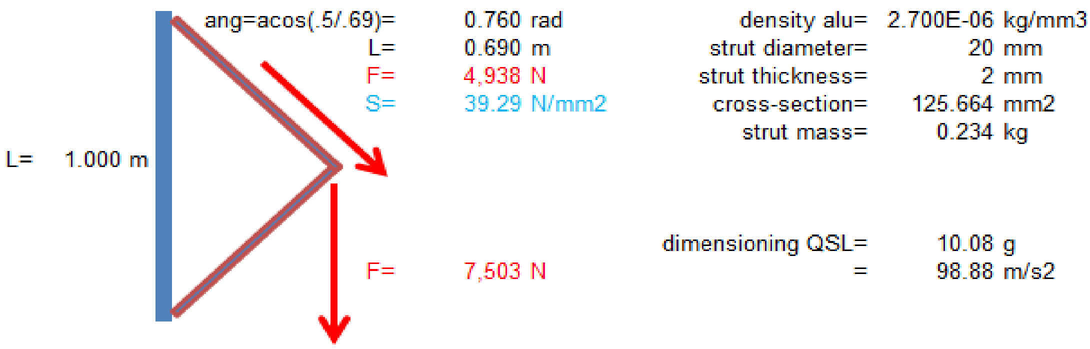


Figure 11-6: LM strut design for tank bipods

11.3.2 Adapter PM-LM

The PM-LM adapter is shown in Figure 10-2 and Figure 11-4 on top of the PM and below the LM struts/connectors and bottom panel (not shown in figure for better visibility).

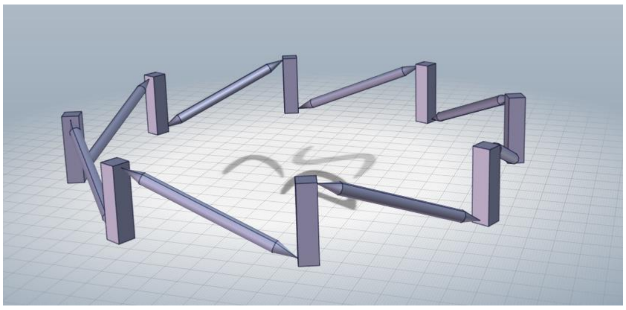


Figure 11-7: Adapter PM-LM

11.3.3 Adapter LM-ERV

The LM-ERV adapter is shown in Figure 11-8 and Figure 11-4 on top of the LM top panel and below the ERV.

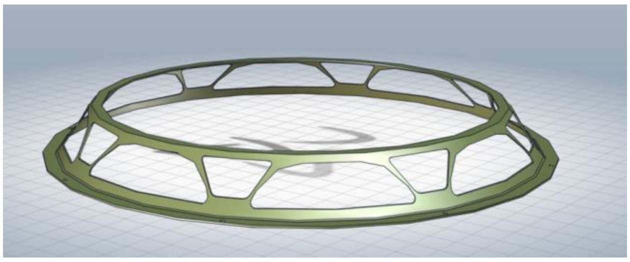


Figure 11-8: Adapter LM-ERV

The design is not detailed further. The main structural requirements it has to fulfil are:

- Provide strength and stiffness during the entire mission
- Provide a smooth, low over-flux upper interface.

11.3.4 ERC

The ERC structural design follows closely the ERC Phootprint design:

- Shapes and size are the same
- Mass is very similar.

Structure:

- Front Shield: a sandwich with two skins of 0.8 mm thick CFRP and a 15 mm thick aluminium honeycomb
- Back cover: 2 mm of aluminium (average value featuring common areas with a lower thickness than 2 mm and local reinforcements thicker than 2 mm)
- Internal structure:
  - A conic shape sandwich, with two skins of 0.8mm thick CFRP and a 5mm thick aluminium honeycomb, which confines the energy absorbing material
  - 6 aluminium panels with lateral stiffeners, which transmit loads at impact and protect the crushable material from crushing on the ground
  - A 2.4 kg ballast mass has been added on the Frontshield structure in order to lower the centre of mass, staying below the maximum mass limitation.

No further design work has been done on the ERC, Figure 11-9 shows the configuration.

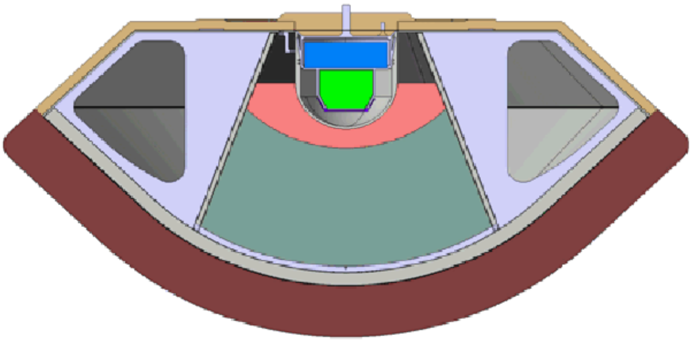


Figure 11-9: ERC structure (from Airbus DS ERC Phootprint design)

11.4 List of Equipment

Table 11-1 shows a list of LM structure equipment

ELEMENT 1 LM		mple Return									
		MATERIAL PROPERTIES						RESULTS			
Create a New Design Sheet / Add a new unit		Sandwich plate		Solid Plate (give only core depth)							Unit mass with margin
	Nr.	sheet material	sandwich: core thickness solid: plate thickness beam: cross section	Material	density	area	M_struct	Material	Maturity	Unit Margin	[kg]
Item			[mm] or [mm²]		[kg/m³]	[m2]	[kg]			[%]	[kg]
EXAMPLE	2	CFRP [0,60,-60]s	30	STEEL	7950	1.00	6.37	STEEL	Modification	10	7.01
Panel lateral (w/o radiators)	6	CFRP [0,45,-45,90]	40			0.88	3.49	sandwich	New dev.	20	4.19
Panel lateral (w/ radiators)	2		4	ALUMINUM	2770	0.88	9.75	ALUMINUM	New dev.	20	11.70
Panel top	1	CFRP [0,45,-45,90]	40			3.74	14.83	sandwich	New dev.	20	17.79
Panel bottom	1	CFRP [0,45,-45,90]	40			3.74	14.83	sandwich	New dev.	20	17.79
Strut/connector longitudinal	8		760	ALUMINUM	2770	1.00	2.11	ALUMINUM	New dev.	20	2.53
IF ring to ERV (octagonal)	1		3	ALUMINUM	2770	7.04	3.20	ALUMINUM	Modification	10	3.52
IF ring to PM (octagonal)	1		5	ALUMINUM	2770	7.04	10.20	ALUMINUM	Modification	10	11.22
Struts tank (4)	16		125.66	ALUMINUM	2770	0.69	0.24	ALUMINUM	New dev.	20	0.29
Fitting tank (1)	4		10	ALUMINUM	2770	0.00	0.20	ALUMINUM	New dev.	20	0.24
Brackets/connectors	1		10	ALUMINUM	2770	0.00	13.50	ALUMINUM	New dev.	20	16.20
Adapter top to ERV	1		10	ALUMINUM	2770	1.00	3.00	ALUMINUM	New dev.	20	3.60
11							121.47			18.9	144.42

Table 11-1: List of LM equipment

11.5 Stiffness Verification

Stiffness verification w.r.t. Proton is discussed in this section. The requirements originate from RD[13]:

- Axial first mode > 35 Hz
- Lateral first mode > 15 Hz.

A FEM has been built containing the following elements:

- PM – represented by a rigid beam with Fregat mass, see Table 11-4
- PM-LM adapter - rigid, mass and dimensions see Table 11-3
- LM - represented by a simplified 3D FEM Figure 11-10, mass and dimensions see Table 11-3
- LM-ERV adapter - rigid, mass and dimensions see Table 11-3

- ERV - rigid, mass and dimensions see Table 11-2 (note: ERV&ERC COG is located at 40.5% of ERV height)
- ERC - rigid, mass and dimensions see Table 11-2
- Tanks – rigid struts, mass and dimensions see Table 11-2 (“mass propulsion”, “propellant mass total”).

The FEM with dimensions is shown/sketched in Figure 11-10

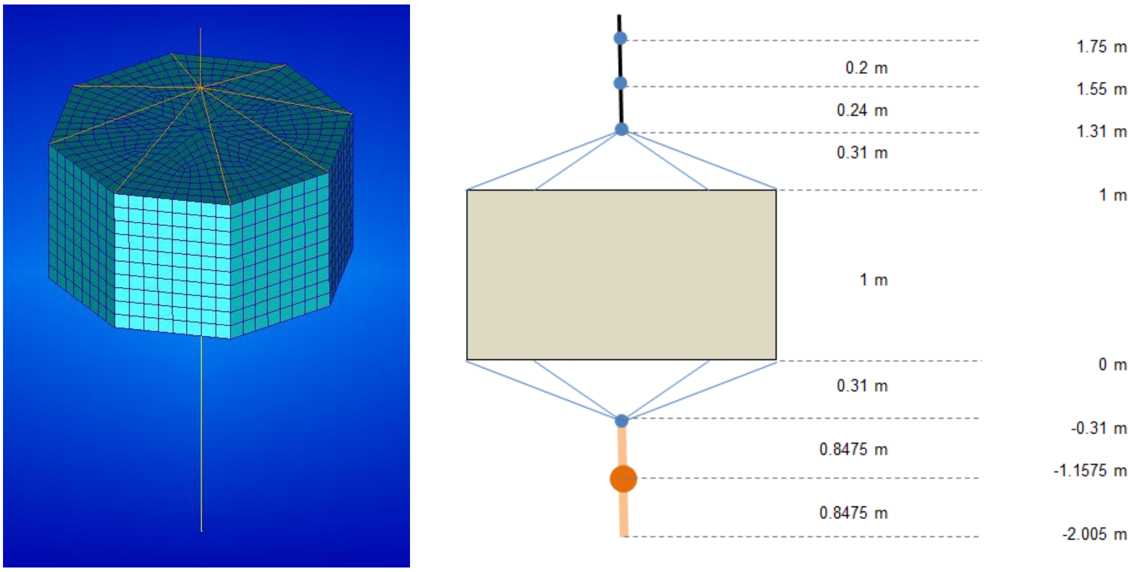


Figure 11-10: Phobos simplified FEM

ERC	mass	kg	35	
	height	m	0.385	
	COG wrt upper if	m	-0.21	
	diameter if	m	0.77	
	frontshield			
ERV	mass	kg	264	
	prop	kg	230	
	height	m	0.65	
	COG wrt lower if	m	0.24	*40.5%*
	diameter if	m	1.6	

Table 11-2: ERV and ERC FEM, structural mass properties

LM	mass	kg	644
	mass structure	kg	120
	mass propulsion	kg	93.46
	height	m	1
	COG wrt lower if	m	
	diameter if	m	2
	# tanks		4
	tank mass	kg	
	propellant mass total	kg	567
	tank cog from lower if	m	
	adapter height ERV	m	0.31
	adapter mass ERV	kg	10
	adapter height PM	m	0.31
	adapter mass PM	kg	20.1

Table 11-3: LM and adapters FEM, structural mass properties

PM	mass	kg	3,165
	diameter if	m	2
	height	m	1.695
	height COG	m	0.8475
	mass structure	kg	650
	mass propellant	kg	2,515
	stiffness axial	N/m	infinite
	bending (unscaled)	Nm/rad	infinite
	stiffness axial	N/m	2.412E+07
	bending	Nm/rad	3.380E+08
	diameter	m	3.3
	height	m	1.5

Table 11-4: PM FEM, structural mass and stiffness properties

Note: values with the colour code are assumptions. Mass values are not always in line with the final mass values of the study. However, the differences in mass values can be neglected with respect to the conclusions derived.

The first eight eigenfrequencies of the Phobos SR system clamped at the Fregat lower I/F are shown in Table 11-5.

Mode	Frequency [Hz]	Description
1	17.43	Lateral aluminium sandwich panel mode
2	28.99	Lateral CFRP panel mode
3	29.09	Lateral CFRP panel mode
4	29.26	Lateral CFRP panel mode
5	95.22	1st global-z bending mode LM

Mode	Frequency [Hz]	Description
6	99.91	1st global-y bending mode LM
7	125.35	Upper panel mode
8	128.13	Lower panel mode

Table 11-5: LM structural FEM main frequencies

As can be observed from Table 11-5, the lowest frequency of the LM of 17.43 Hz is related to the aluminium sandwich panel, which is above 15 Hz as required. The modes of the CFRP panels are at around 29 Hz.

As the axial and bending stiffnesses of the Fregat in the particular configuration for Phobos-SR were not available, the global behaviour could not be checked. What is learned from this analysis is that the LM global bending modes are very high, as well as the longitudinal modes, so no problem from the LM is expected once coupled to representative PM and ERV FEMs.

At a later stage it needs to be verified whether these Proton lateral and axial stiffness/frequency requirements are met.

As far as only the LM is concerned, the global stiffness is sufficient.

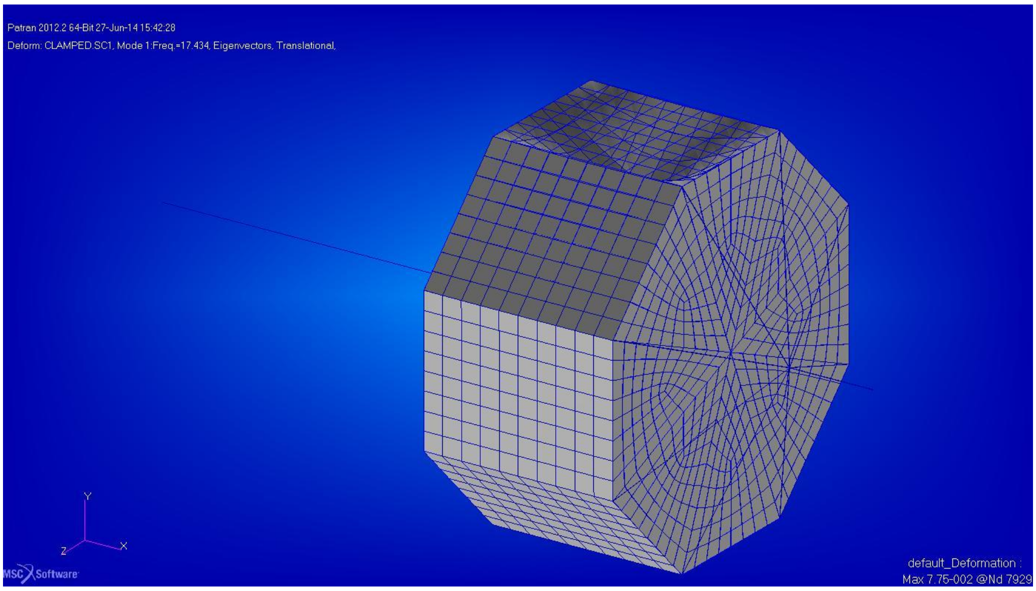


Figure 11-11: LM lateral aluminium sandwich panel mode

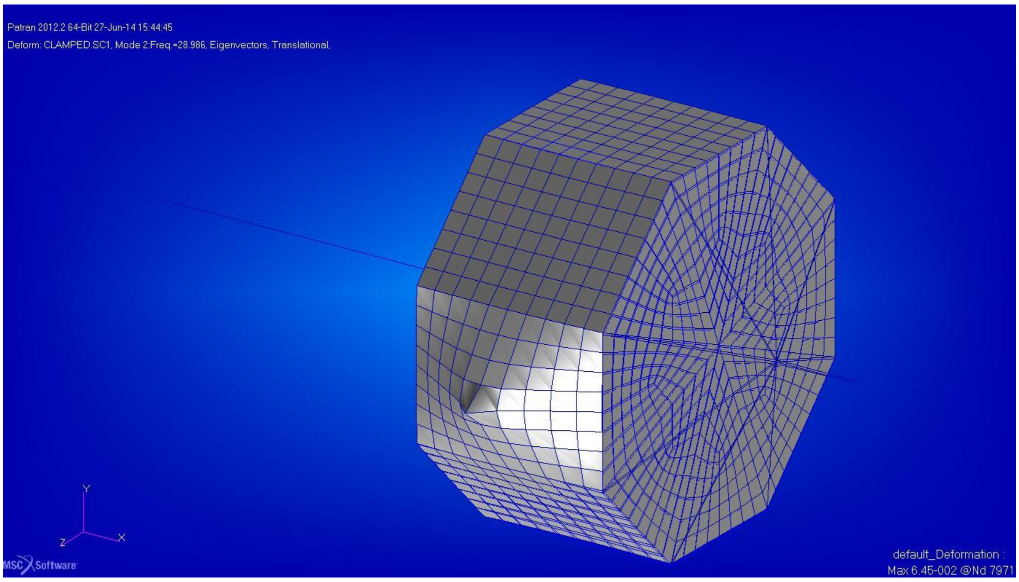


Figure 11-12: LM lateral CFRP panel mode

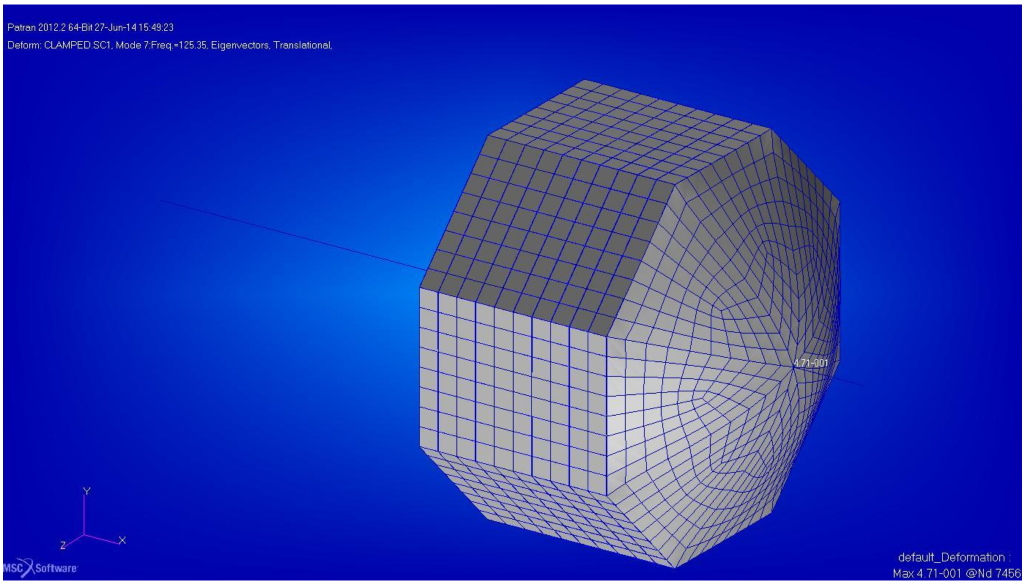


Figure 11-13: LM upper panel mode

11.6 Technology Requirements

No particular technology development is required for the structural components of LM and the PM-LM adapter below, and LM-ERV adapter above the LM.

12 ROBOTICS

12.1 Requirements and Design Drivers

SubSystem requirements		
Req. ID	**STATEMENT**	**Parent ID**
MI-10	The mission shall return approximately 100g of loose material from the surface of Phobos	
MI-220	The mission shall perform 3 types of surface operations in order to fulfil requirements in the Science Requirements Document: - Sampling point selection and characterisation - Sample acquisition and transfer to ERC - Post-sampling science measurements	
MI-260	Once landed, the mission shall allow the Ground to select the sampling location within the sampling tool range	
MI-270	The mission shall provide the possibility to the Ground to check that the collected sample is suitable before transfer to the ERC	
MI-280	The mission should implement on-board automatic procedures to perform contingency sampling and lift-off operations in case of communication failures with the ground	
CO-30	The Composite shall provide single point failure tolerance. Redundancy concepts shall be considered to minimise consequences of single point failures *C: any deviation with respect to this requirement shall be identified and justified*	
CO-60	In the Composite design, only technologies that can be assumed to be at TRL 5 at the start of the mission implementation phase shall be considered when defining the mission architecture	
LM-60	The LM shall allow the sampling, transfer to ERC and sealing of the sample	

Table 12-1: Subsystem requirements

Note that the following chapters refer to the ESA sampling chain solution described in chapter 24. However in case the IKI sampling chain solution would be used (see chapter 25), it is considered at this preliminary stage that a similar robotic arm would be used to transfer the IKI sample container to the ERC.

12.1.1 Robotic Arm Design Description

The robotic arm is required after landing on Phobos in order to:

- Transfer the sampling device from its stowed position to close proximity with the surface
- Exert the necessary force required during sampling
- Place the sample container into the ERC with the required forces.

The arm needs to function in the environment of Phobos and be able to take a sample from an area beneath the lander.

12.2 Baseline Design

12.2.1 Robotic Arm Description

A kinematic analysis has been performed to design the kinematic structure of the arm.

The arm will be constructed with two limbs of 1.2meter to have sufficient reach to touch the surface and access the ERC. 4 degrees of freedom are needed, these will be realised by 3 pitch joints (J1, J2 and J3) which give the robot arm the ability to transfer a sample from the ground to the ERC and a roll joint (J0) which gives the arm the ability to access a wider sampling area and also provides more freedom for the location of the arm on the lander platform. The design with two long limbs of a similar length provides the most compact configuration when stowed, without increasing the number of limbs and joints.

The main characteristics of the design are:

- 4DOF Robotic arm
- Two limbs: L1=L2=1.2m
- Mass = 14.3 kg
- Sampling area: $A = 3.2m^2$
- Sampling tool is attached to arm at the start of mission
- Rotary bristles sampling tool

Figure 12-1: Sampling area

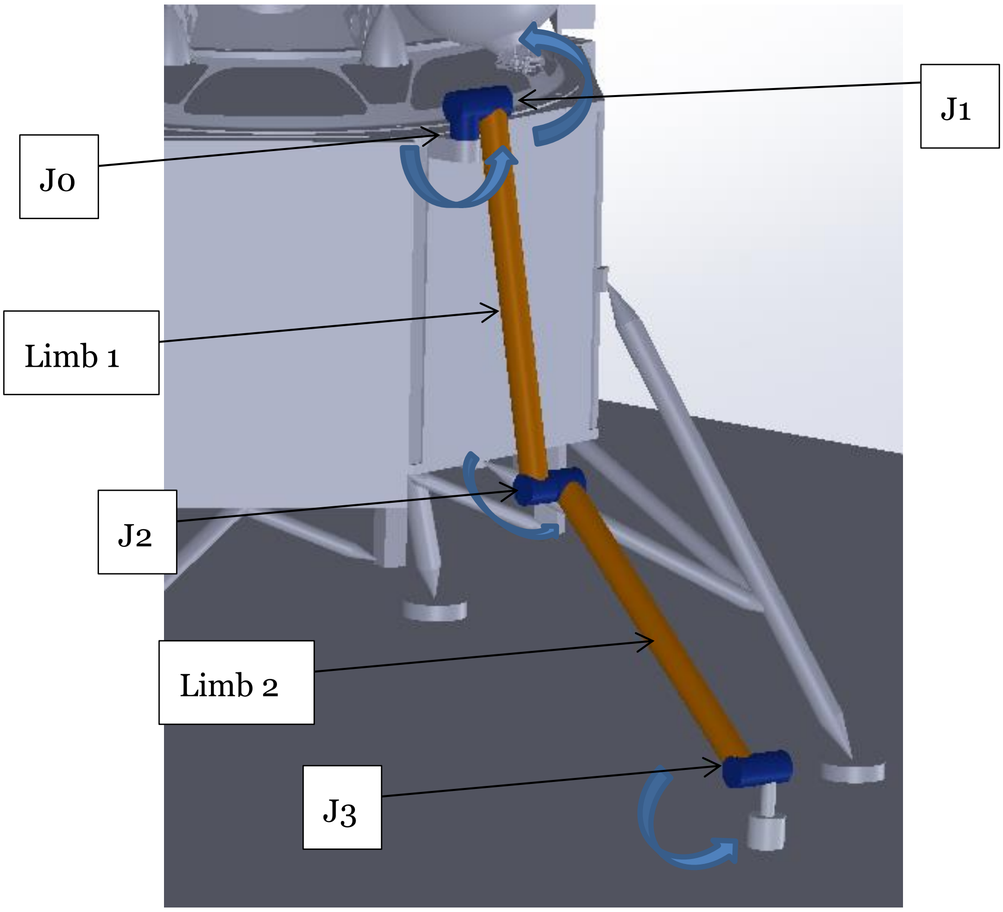


Figure 12-2: Robotic Arm components

12.2.1.1 Joint Torque Requirement

This section addresses different loading cases for the joints to determine the torque capability for each joint.

The three loading cases in this mission are 1) sampling (10N, from Phootprint system studies), 2) storing the sample in the ERC (40N, from Phootprint system studies) and 3) gravity load case

Sample acquisition case

- The arm should exert 10N towards the surface at any position the arm could reach.
- The worst case is assumed to be fully extended. This case is illustrated in the figure below.

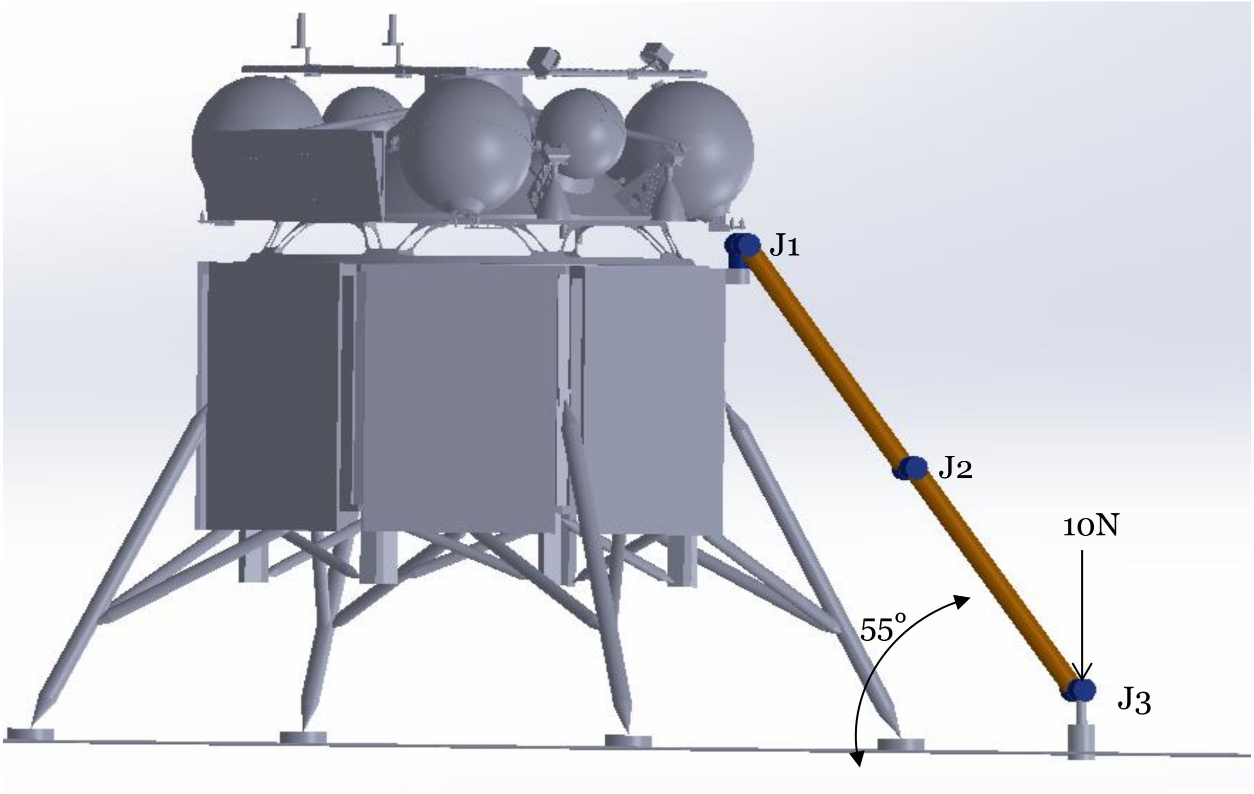


Figure 12-3: Sampling acquisition case

The required torque in J0 is negligible because the direction of J0 is perpendicular with the force exerted at Phobos.

The required torque in J1 is $M = F * x = 10N * 2.4 * Cos\ (55) = 13.76\ Nm$

The required torque in J2 is $M = F * x = 10N * 1.2 * Cos\ (55) = 6.88\ Nm$

The torque in joint 3 is negligible because it is in line with the direction of the force and the sampling tool.

Returning sample to ERC

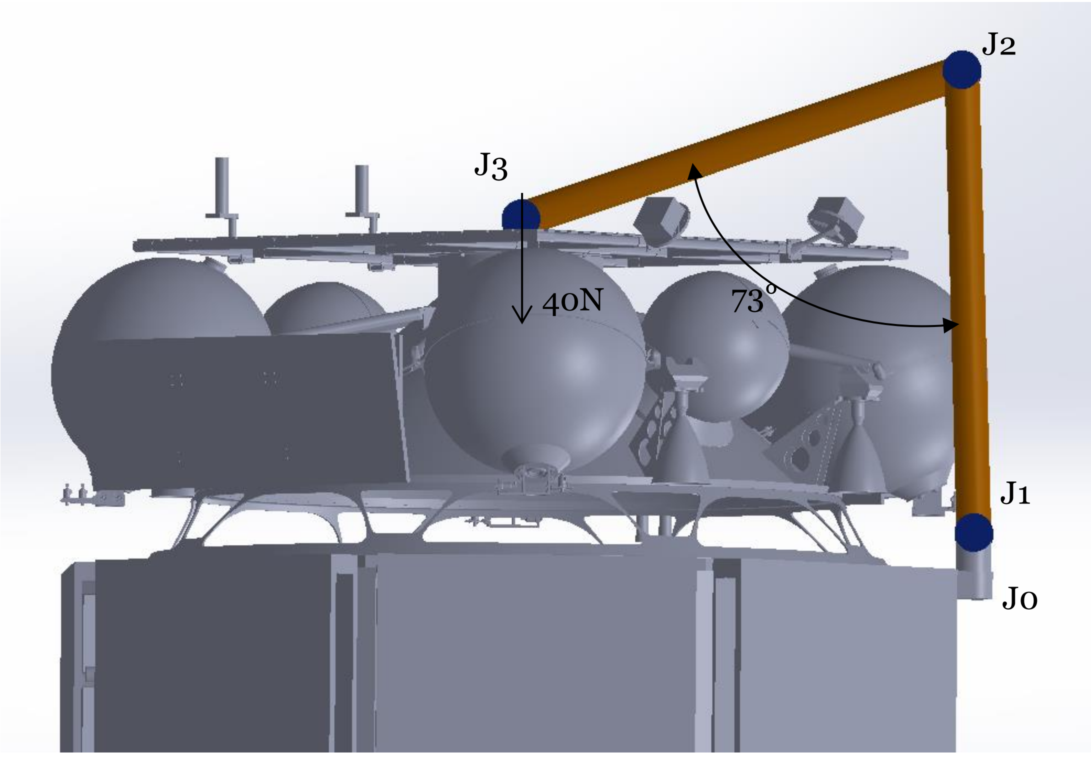


Figure 12-4: Returning the sample to the ERC

The required torque in J0 is negligible because the direction of J0 is perpendicular with the force exerted at the Phobos.

Joint 1 has no torque to apply because the arm is parallel with the direction of the applied force.

The required torque in J2 is $M = F * x = 40N * 1.2 * Cos\ (17) = 45.9\ Nm$

The required torque in J3 is nil because it is in line with the force.

Weight load case

The arm shall hold its position unpowered for extended periods. The worst case is considered when the arm is fully extended from the lander and is perpendicular with respect to the Phobos gravity vector. The mass of the arm, limb, joints and payload is considered due to the Phobos gravity vector assumed to be 0.0057m/s^2.

Joints:

To estimate the weight of the joints it is valuable to refer to the scalability analysis which is reported in the document "Preliminary Design Document and Scalability Analysis" RD[14] from the Dextrous Lightweight Arm for Exploration (DELIAN) project. Four families of joints have been identified, covering all DELIAN application scenarios. They have been preliminarily sized in terms of torque, mass and dimensions.

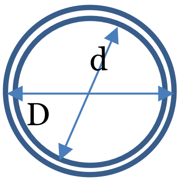
Torque and mass are as follows:

- 0 - 4 Nm, 0.356 kg

- 5 – 12 Nm,   0.438 kg
- 13 – 40 Nm,   0.585 kg
- 40 – 75Nm,   1.04 kg

Limbs:

The limbs have to be strong enough and therefore a rough calculation will provide an estimation of the weight of one limb. The limbs will be made from the light aluminium beam with an outer diameter of 44mm, 1mm thickness and have to withstand a bending moment (Mb) of 45 Nm.



Therefore $\sigma = \frac{Mb*y}{I}$

$$I = \frac{\pi * R^4}{4} - \frac{\pi * r^4}{4} = 3.123 * 10^{-8} m^4$$

Figure 12-5: Limb section

σmax = $3.24*10^{10}$Pa

σalu = $7 * 10^{10}$Pa

The aluminium thin-walled beam is able to withstand the small forces produced by the payload.

<u>Gravity load case</u>

The gravity load case has two aspects:

- Load on Phobos during mission
- Load on Earth during functional testing

First some common values are computed.

The limbs are made of hollow cylinders with an outside diameter of 44mm and inner diameter of 42mm. The length of the limbs is 1.2m and the density (ρ) of the aluminium is 2800kg/m^3

$$V = \frac{\pi * (D^2 - d^2)}{4} * l = 1.621 * 10^{-4} m^3$$

$$m = V * \rho = 0.453 kg$$

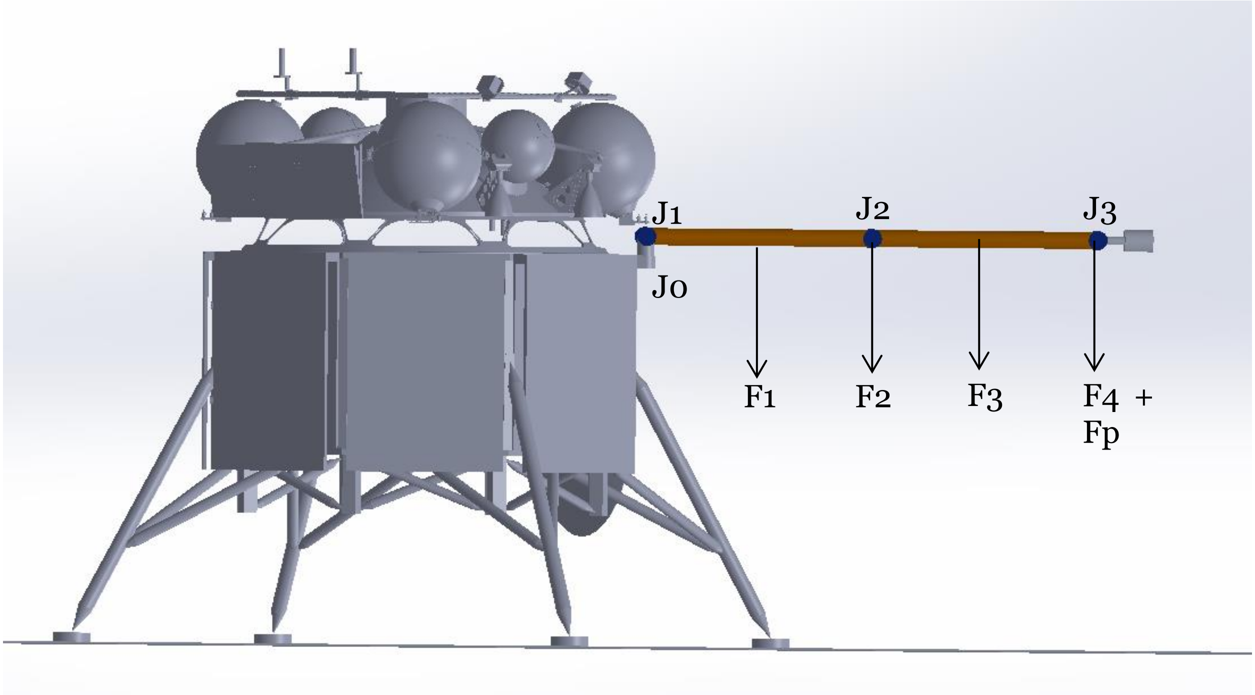


Figure 12-6: Gravity load case

m1	= Mass limb one	=0.453 kg
m2	= Mass Joint two	= 1.014 kg
m3	= Mass limb two	=0.453 kg
m4	= Mass joint three	= 0.438 kg
mp	= Mass Payload	= 4.1 kg

$$T1 = m1 * g * \frac{l}{2} + m2 * g * l + m3 * g * \frac{3l}{2} + m4 * g * 2l + mp * g * 2.1l$$

$$T2 = m3 * g * \frac{l}{2} + m4 * g * 1l + mp * G * 1.1l$$

$$T3 = mp * g * 0.1$$

Phobos gravity load case:

g = Phobos gravity= 0.0057 m/s²

The required torques for the joints are:

T1=0.0780Nm

T2=0.0359Nm

T3= 0.002337Nm

The direction of joint 0 is perpendicular to the force direction which means this load is not affecting the joint. Joint 3 does not have to apply a torque to be stable in this load case.

Earth testing load case:

When replacing g of Phobos with 1g on Earth (9,81m/s²) the torque will be:

T1=134.27Nm

T2=60.91Nm

T3=4.02Nm

These values are much higher than the torque needed by any other case. In order not to size the whole arm just for the ground testing case, an adapted testing scope is introduced.

In this situation the payload during testing is replaced with a lighter model (0.25 kg) so that the highest torque is 39Nm (compatible to the worst torque of other cases). Other approaches should also be used, to be further defined in the development and validation plan (e.g. compensation devices, 2D test setup, etc...)

Conclusions and Summary

The worst case regarding torque demand can be found in joint 2 during insertion of the sample into the ERC. The arm is not able to operate in Earths 1G environment with the actual payload attached. The Robotic arm tests should be done first with a lighter payload and secondly with the actual payload in a simulated 0G environment. Horizontal 2 dimensional tests are recommended here because joint 0 is there for the third dimension and is never loaded by high torques. This concludes the whole system could be tested in two phases.

The following table summarises the loads on the joints in the different cases.

	Load	Joint 0	Joint 1 (Nm)	Joint 2 (Nm)	Joint 3 (Nm)
Insert into ERC	40N	-	-	45.9	-
Sampling operation	10N	-	13.76	6.88	-
Gravity load torque	0.0057m/s²	-	0.075	0.035	0.002
Gravity load torque(normal payload)	9.81m/s²	-	134.27	60.91	4.02
Gravity load torque(0.25kg payload)	9.81m/s²	-	39.09	11.05	0.245

Table 12-2: Load cases summary

The next table shows only the maximum torques of the different situations on Phobos and on Earth with normal and reduced payload. This table is a tool to keep an overview of the masses in different scenarios. The masses of the components on Earth are based on the previous MMSR_A5 study[RD[16]]. This study assumes testing with payload in 1g environment.

		Arm compatible with nominal payload during Earth testing		Arm compatible with reduced payload during Earth testing	
Joint	Maximum torque Phobos (Nm)	Maximum torque on Earth	Mass (kg)	Maximum torque Earth	Mass (kg)
Joint 1	13.76	134.27	2.5	39.09	0.585
Joint 2	45.9	60.91	2.5	11.05	1.014
Joint 3	0.002337	4.02	1.5	0.0245	0.438
Limb 1	-	-	2	-	0.453
Limb 2	-	-	1.5	-	0.453

Table 12-3: Maximum Loads

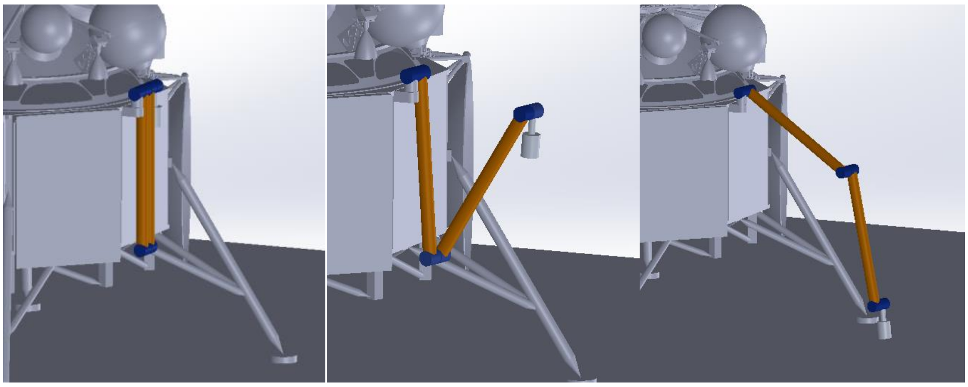
12.2.2 Sample Acquisition Sequence

The Robotic arm is stowed along a side panel of the Lander vehicle.

Two hold-down brackets hold the robotic arm in place during launch and transfer to Phobos. Once the lander is on Phobos these brackets can release the Robotic Arm and the arm can start with its operations.

There are two sampling modes foreseen, in order to have redundancy in case the communication with the ground station is lost.

In the first mode the lander takes a picture of the surface beneath the arm and sends it to Earth. Ground control decides on this information where to take the sample. The Arm is instructed to travel from the stowed position to the surface of Phobos. The sampling tool can start the brushes just before the arm reaches the soil and the arm can continue its descent. After the sampling tool has collected the sample and this is verified by means of its sensor, the arm moves upwards to the ERC. During this transfer the sampling tool, described in the sampling chain section, opens up and prepares itself to deliver the container to the ERC. The arm has to exert a force of 40N into the ERC to lock the container into the ERC. The arm returns to the original stowage position after separating from the sample container (to free space for the ERV to take-off). This sequence of operations is illustrated in Figure 12-7.



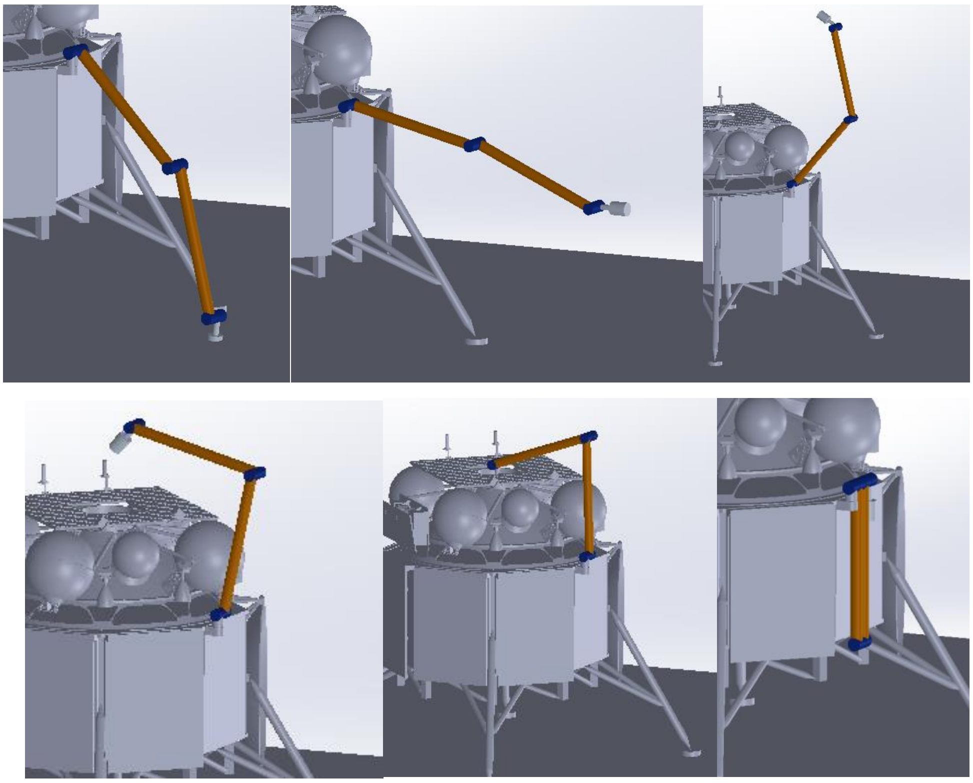


Figure 12-7: Sampling chain sequence

In the second mode, the Arm and Sampling Device will perform the same sequence autonomously if the communication with the ground station is lost. The location of sampling will be one pre-recorded in the system.

These modes and the associated sequences are described in the chart Figure 12-8 which is extracted from the RD[15] document.

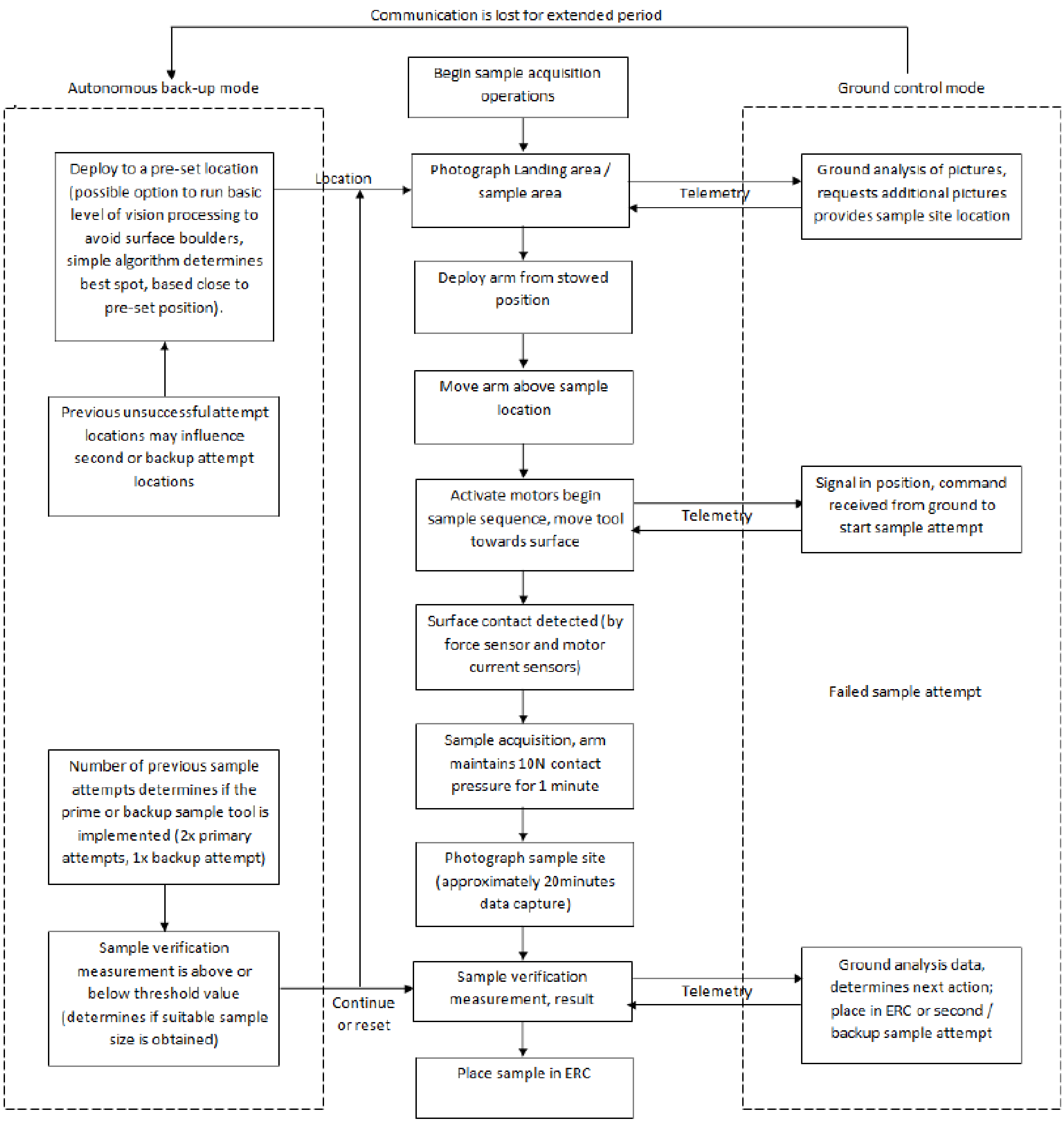


Figure 12-8: Operations diagram

12.2.3 Motors

The choice of the actuator has been inspired from the DELIAN technology development. The trade-off leading to the choice of actuator is found in the document "Preliminary Design Document and Scalability Analysis" RD[14]. A summary of the conclusions is presented hereafter.

12.2.3.1 Summary

From a mechanical point of view, brushless DC torque motors appear as the best candidate for lightweight joint design as required in DELIAN. However the increased control complexity and the higher complexity of harness with respect to brushed motors needs to be considered in the mass and power estimation. When considering the length of the harness and the mass of control electronics, brushed DC motors result the best candidate for the DELIAN application of long sampling arm. Lifetime assessments reported in the annex of RD[14] demonstrate that even in vacuum environment such a motor fulfils the lifetime requirement of short missions.

12.2.4 Harness

Whether the harness is internal or external to the arm structure, the torque necessary for bending and twisting of cables can be substantial at the low temperatures expected on Phobos. Especially with high rotation angles (>90° deg) the bending torque resistance of normal electric wires increases significantly.

Flex printed circuit harness is a solution for minimising resistive torque. Referring to annex III of the DELIAN arm RD[14] the resistive torque of the Flex print is only 0.03 Nm for 24 wires. This torque is negligible with respect to the torque used in the joints.

With respect to mass, in this phase of the mission design, the mass of the harness is conservatively estimated as 10% of the total robot mass.

12.2.5 Thermal Protection

The thermal protection is estimated as 5% of the total robot mass.

12.2.6 Hold Down and Release Mechanism (HDRM)

The robotic arm is held against the landers wall by two HDRM locations. The first one is located at the end of limb 1 close to joint 2. The second one is located at the end of limb 2, close to joint 3. The HDRM is described in the Phootprint document RD[15].

12.2.7 Mass Estimation

The following table presents all the mass data of the Robotic Arm and its total mass.

The mass is calculated for two scenarios. In the first scenario, the arm cannot be functionally tested on Earth with full payload.

In the second scenario the arm is able to do full sampling tests on Earth. This implies that the joints and structure have to take the weight of the system into account. Therefore the whole system ends up heavier. The mass of the second scenario refers to the MMSR_A5 study RD[16].

Tool	Mass (reduced payload on Earth) [kg]	Mass (full Payload on Earth) [kg]
Joint 0	0.585	2.5
Joint 1	0.585	2.5
Limb1	0.453	2
Joint 2	1.014	2.5
Limb2	0.453	1.5
Joint 3	0.438	1.5

Total bare	3.528	12.5
Harness + thermal protection. (15%)	0.529	1.8
Total	4.057	14.3

Table 12-4: Mass Estimation

12.3 List of Equipment

The equipment below is also listed in the mechanisms chapter (only the robotic arm is to be considered in the present chapter).

Element 1	Lander			MASS [kg]			
Unit	Unit Name Click on button above to **insert new unit**	Part of custom subsystem	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
1	Landing gears		4	5.6	To be developed	20	26.9
2	Sampling tool		1	4.1	To be developed	20	4.9
3	APM+APME+HDRM		1	12.5	To be modified	10	13.8
4	SA deployment mech. + HDRM		6	5.0	Fully developed	5	31.5
5	Robotic arm		1	14.3	To be developed	20	17.1
-	Click on button below to insert new unit						
SUBSYSTEM TOTAL			**5**	**83.3**		**13.1**	**94.2**

Table 12-5: Equipment list

12.4 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Included in this table are:

- Technologies to be (further) developed
- Technologies available within European non-space sector(s)
- Technologies identified as coming from outside ESA member states.

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
Robotic arm	Transferring a sample from a moon to the ERC.	Selex TRL4 by 2016	Yes	See ESA MREP DELIAN activity information

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13 MECHANISMS (EXCLUDING THE SAMPLING TOOL)

Important note: this chapter is complemented by the chapters describing the two different sampling chains solutions proposed by ESA (chapter 24) and IKI (chapter 25). In particular that is the reason why the sampling tool is not described in this chapter, because the sampling approaches are very different and are described in the previously mentioned chapters.

13.1 Requirements and Design Drivers

13.1.1 ERC Spin and Ejection Device

For the ERC separation, spin and ejection, the following design drivers are applicable:

- The ERC shall be spun and ejected
- The Spin and Eject Device (SED) shall be able to support the ERC probe from launch to ejection.

13.1.2 Sample Container

The sample container will store the samples collected by the sampling tool while attached to the robotic arm during the sampling operation. Afterwards it will be transferred into the ERC. Its requirements are closely related to those of the sampling tool. The sample container:

- Shall be quasi-hermetically closed
- Shall remain attached to the robotic arm during launch and till the end of sampling
- Shall guarantee a reliable samples collection and hold of samples
- Shall be able to measure the amount of samples collected during the sampling operation
- Shall be transferred to the ERC by the robotic arm after sampling
- Shall remain firmly attached to the ERC from its transfer till the end of re-entry into Earth.

13.1.3 Landing Gears

- Shall ensure a stable landing position on Phobos surface
- Shall be compatible with Phobos surface characteristics (mechanical strength, roughness, inclination, very low gravity etc.)
- Shall tolerate residual landing speed (vertical and transversal), attitude angle and angular speed of the S/C
- Shall limit as much as possible landing shocks and accelerations
- Shall dissipate as much as possible the kinetic energy of the S/C before landing.

13.1.4 High Gain Antenna Pointing Mechanism (HGA APM) and Hold Down and Release (HDRM)

For the HGA, the following functions have to be fulfilled:

- Restrain at launch with hold down and release mechanisms
- Deploy once launched
- 2 DOF pointing function is required
- withstand the landing loads when landing on Phobos.

13.1.5 Solar Array Deployment Hinges and Latches, and Hold- Down and Release Mechanism (HDRM)

With regards to the solar arrays, the following drivers have been considered:

- The SA hinges shall be able to withstand the landing loads on Phobos, and the firing of the PM thrusters during orbital manoeuvres
- SA surface of 10.5 m^2, arranged in 5 wings, each one made of 2 foldable panels, with each wing having a mass of about 7 kg.

13.2 Assumptions and Trade-Offs

This paragraph will summarise some general assumptions for the mechanisms used in this study.

- Phobos environment has a temperature between 100 and 300 K. Mechanisms with bearings and gears, if lubricated with grease, must probably be provided with heaters in order to keep the temperature between -40 and 60 degC during operation
- The Earth Return Vehicle (ERV) separation mechanism from the Lander will be provided by the Russian space agency and is therefore not described here.

For the Landing Gears, some specific assessments have been done and will be presented in the following dedicated section.

13.2.1 Landing Gears

In the following sections, the touch-down stability conditions, forces and accelerations on the Landing Module with ERV and ERC on top are evaluated with simplified assumptions. The objective is not to derive a quantitative estimation, but to provide an understanding of the effect of the various parameters in order to later develop guidelines to mitigate possible critical behaviours.

Evaluations are done using a 2-dimensional model, and energy and momentum balance equations. A common assumption for all the evaluations is that the crushable material presents a flat or plastic force-displacement relationship. A real crushable material has a more complex behaviour; nevertheless the assumed threshold force can be regarded as an “equivalent” value from an energy dissipation point of view. It is assumed for simplicity that the angular speed of the S/C is negligible. It is finally highlighted that the angle of the S/C when approaching the soil shall be compatible with the envelope of the deployed solar array, in order to avoid any clash.

13.2.1.1 First legs touch-down stability

Some rough investigations have been done about the stability of the lander during the touchdown.

Very simplified assessments can be firstly made on the geometry of the legs (constrained by the launcher envelope and ground clearance) and the flight parameters before the first leg touches down. For simplicity, a 2-dimensional model is used, no initial rotational speed and the S/C is assumed to land on two legs.

The main condition for the first touchdown is that the S/C does not rotate in the direction to flip-upside down. The further hypothesis that the touch-point will not be able to slide sideward is also introduced.

The condition which drives the evaluation is that the translational momentum vector must have an arm to the touch-point, which produces a rotational momentum to roll the S/C so that it will later touch with the second pair of legs.

From the momentum equation we obtain:

$$\frac{v_{x0}}{v_{y0}} < \frac{1 - \tan(\alpha_r)\dfrac{h\sqrt{2}}{r}}{\tan(\alpha_r) + \dfrac{h\sqrt{2}}{r}}$$

Where:

h: height of the CoG;

r: radius of the footprint of the legs;

α_r: angle between the S/C and the ground, due to attitude and surface slope combination;

v_{xo}, v_{yo} : speeds of the S/C along the surface and orthogonal to it;

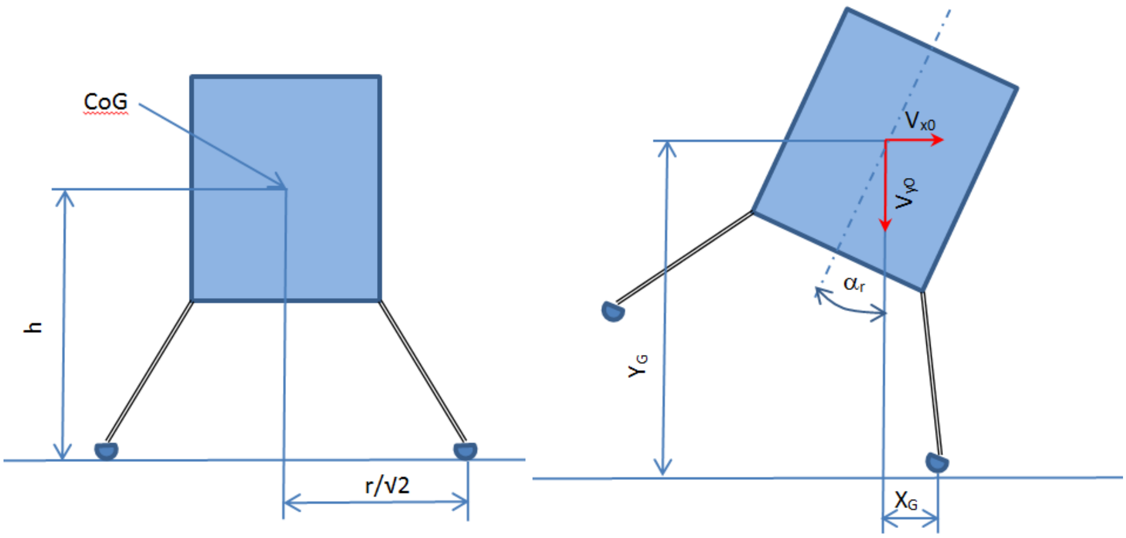

Figure 13-1: Sketch of the S/C approaching touchdown and main symbols

The condition says for instance that with $h/r= 1$, and $\alpha= 25$ *deg*, the max ratio between the horizontal and vertical speed is $v_{xo}/v_{yo} = 0.18$. The relationship highlights that the higher the CoG wrt the legs footprint (h/r), and the higher the angle of the S/C at touchdown (α_r), the smaller must be the residual horizontal velocity.

13.2.1.2 Peak force for one-leg touch-down

From the translational momentum equation, it is also possible to assess the max impulse force which a leg will undergo during touchdown. The worst case occurs when the first landing is on only one leg, and the translational momentum vector is directed towards the touchdown point.

Considering a speed of 0.7 m/s, and a mass of 1400 kg, the impact impulse to completely stop the motion is about 1000 Ns. If a crushable leg with *2000 N* of threshold force is used, the impact will occur in *0.5 s* and the acceleration the S/C will experience is $1.4\ m/s^2$. The crush length needed results in about *0.18 m*.

Even if these evaluations are extremely simplified, a first guess of the order of magnitude of key parameters can be obtained.

13.2.1.3 Second leg pair touch-down stability and hold-down thrust

After the first legs impact the soil, the S/C starts to rotate till the second pair of legs reaches the ground. The rotation speed is crucial to understand if the following motion will be stable of produce a flip-over of the S/C. Hold-down thrusters must apply a force along the S/C axis to reduce the rotational momentum and produce a stable landing.

The calculation of the thrust force and application time is the purpose of the following assessment.

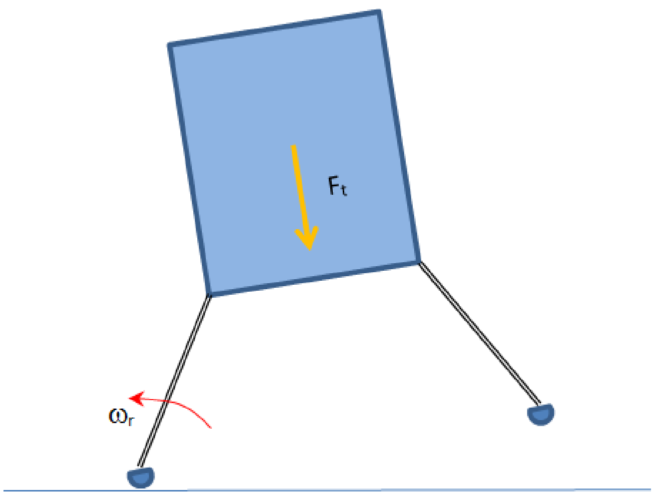

Figure 13-2: Sketch of the S/C rotating around the second contact point (left). The thrust force is shown

It is assumed that the second impact will occur on two legs simultaneously. The worst case happens when the translational momentum have the largest arm w.r.t. the first touch-down point. In this situation, the least energy is absorbed by the first impact, and the most is converted into rotational energy (or momentum) around the first impact point.

Two scenarios can be evaluated for the second impact.

In a first one, it is assumed that the crushable provides no dissipations, and the rotation kinetic energy and momentum has to be reduced by the thrusters only. To give a numerical example, v_{xo} in assumed *0.2 m/s*, v_{yo} *0.6 m/s* and a mass moment of inertia of *1300 kgm²*. After the first crash, a residual rotational momentum of *1050 Nms* can be estimated, with a rotational speed of about *0.11 rad/s*. The thrusters are supposed to act after the second pair of legs touches the soil. Since the dissipation is neglected, the rotational momentum is conserved during the second impact. The necessary impulse from the thrusters is about *750 Ns* (*1.4 m* arm wrt the rotation point). This means that a force of 40 N for example should be applied for *19 s*, meanwhile the S/C makes a rotation of roughly *55 deg (1 rad)*.

In a second scenario, the rotation is assumed to stop completely due to the dissipation of energy occurring in the second crush. With a rotation arm of 2*.8 m*, the impulse needed to balance the rotational momentum is *380 Ns*, about *200 Ns* per leg. This value is of the same order of magnitude as the one estimated for a complete dissipation during the first crush in one leg. It drives therefore comparable requirements on the crushable material.

The two scenarios represent however limited situations which are in any case quite unrealistic. However, the actual behaviour will be something between these scenarios.

13.2.1.4 Flat touch-down on 4 legs

In a perfect flat landing of 4 legs, an estimation of the maximum force per leg, the acceleration and crush-length is estimated.

The initial kinetic energy is 250 J with *0.6 m/s* speed. A threshold force for the crushable legs of 2000 N is assumed. Since the geometry (inclination) of the legs is not considered here, this force must be regarded as a "vertical equivalent". The energy balance gives an estimated crushed length 0.032 m, and a maximum acceleration of 5.7 m/s^2. The crush period lasts 0.1 s, and the impulse per leg is 210 Ns.

It appears that this scenario represents the worst case in term of accelerations, and therefore loads on the rest of the S/C. A possible mitigation can come from reducing the equivalent threshold force on the crushable material (allowing a higher crush length in the preceding scenarios) and/or using elastic deformations to absorb impact energy more smoothly. Any elastic energy will be released back as kinetic energy, therefore a re-bounce of the S/C will occur. This motion has to be controlled by the hold-down thrusters.

A trade-off of these parameters is out of the scope of the actual study. The investigation nevertheless was able to point out the main phenomena involved and their effects.

13.2.1.5 Multi-body simulations

In order to have more confidence about the modelling approach, some simple models have been created in a Multi-body simulation software (DCAP).

The main objectives were to find a condition for a dynamically stable touch-down and understand the main effects of the model parameters, like legs footprint, soil and crushable material properties, level of hold-down thrust etc. The most interesting outputs which have been considered were the vertical touch-down force needed, the max accelerations, and the necessary crushable length.

At this very early level of development, many assumptions and model simplifications have to be made. The results are not intended to be accurate, but have to be regarded merely at the level of order of magnitude evaluations and trade-offs.

For the sake of simplicity, the analysis which is shown here does neglect any elastic deformation, that is, it is assumed that all the energy entirely is dissipated by the crushable material or the sandy ground. With these hypotheses, even if not completely realistic, we can calculate worst case accelerations and loads on the legs.

On the other hand, the implementation of a more realistic elastic behaviour of the legs and of the ground needs several more assumptions over unknown parameters, which lead eventually to results which are anyway not relevant.

A vertical speed of 0.6 m/s and 0.2 m/s horizontal (conservative) is used, with an initial inclination of the S/C of 25 deg. The simplified assumptions of the simulation include: no initial rotation rate, absence of bouncing effects, an hold down-force of 4X10 N for 15 s and a crushing force of 1000N per leg.

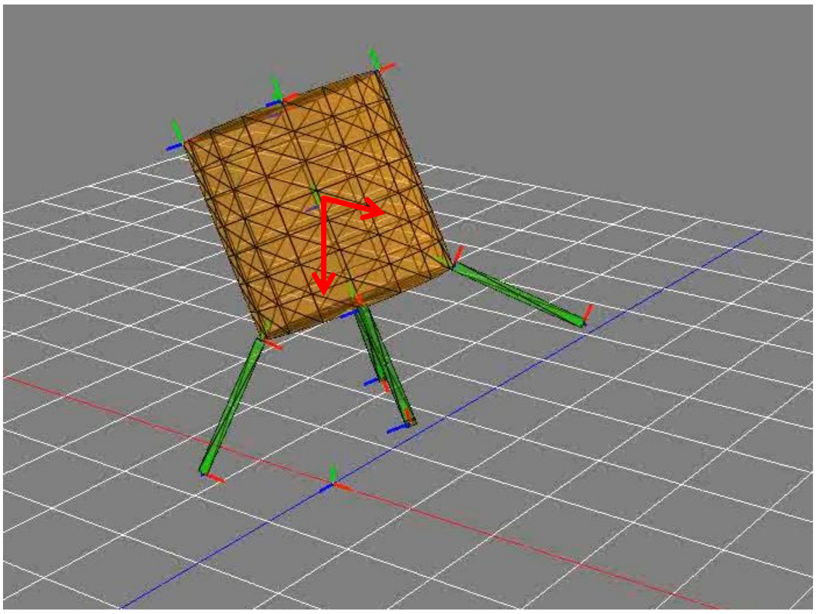

Figure 13-3: S/C approaching landing in DCAP Multi-body model. Vectors of initial speeds are shown in red

The following graphs show the speeds at the S/C CoG. After the first crash, which absorbs part of the kinetic energy, the rest of energy is transferred into rotation and finally dissipated by the second crash. The rotational speed decreases during the first and second crash due to the dissipation occurring on the first two contact points, which are sliding in the actual model simulation.

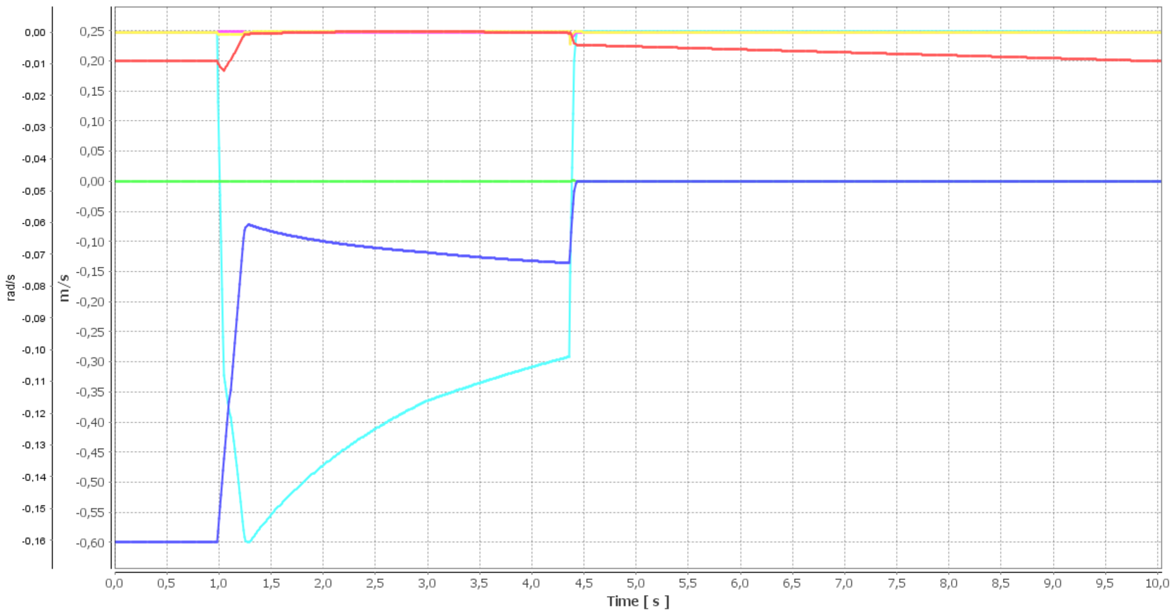

Figure 13-4: S/C velocities at CoG: v_{xo} (red), v_{yo} (blue), and angular speed ω_r (light blue). The steep variations during the two crashes are visible

Accelerations on the CoG, translational and rotational, have peak values which depend strongly on the assumed parameters, like the threshold force of the crushable legs and the stiffness. At the moment they are merely kind of an educated guess. The first touch-down produces vertical acceleration values between *1* and *1.5 m/s²*, for a time of *0.5* to *0.75 s*. The peak when the second pair touches down is significantly lower.

Rotational accelerations are important since they induce further loads and stress on deployed appendices, like the Solar Array or booms for the thrusters. A more accurate representation of the behaviour of the legs is needed to assess the range of rotational accelerations during impact, especially elasticity and energy dissipation shall be modelled along different directions.

In general, and within the scope of this simple study, the behaviour and the results are comparable to those obtained with analytical models based on energy and momentum balance.

13.2.1.6 Conclusion

The landing forces, accelerations and stability conditions have been assessed with different simplified approached under three main conditions:

- Crush on two legs and stability in worst case of attitude and ground inclination
- Crush on one single leg
- Rotation after the first pair of legs crush and conditions on the hold-down thrust
- Flat touch-down on 4 legs.

A multi-body model has been created in DCAP, and some simulation accomplished to gain some more confidence.

The main results can be summarised with the following observations:

- The S/C final attitude, vertical and horizontal speeds w.r.t. the ground determine the level of momentum and energy to be absorbed by the legs, and therefore set the level of criticality for the crushable legs
- The energy absorption characteristic of the crushable material, and the soil, are important for the reduction of the load on the structure and the acceleration that all the S/C is subjected. Particularly critical are those mechanisms which support deployed elements, like the APM and the hinges and latches of the SA
- A hold-down thrust of few tens of N (compatible with the baseline thrusters accommodation) is needed to reduce the rotational speed of the S/C and avoid flip-over, especially in those particular conditions where the speed along the direction of the ground is particularly high. In particular for the preliminary analyses performed during this study a vertical force of 4x10N for about 15s has been retained.
- The structural design of the legs has to trade also the elastic stiffness levels, especially w.r.t. load components orthogonal to the main line of action of the crushable material. In general, the lower the stiffness is, the lower is the impact force, but the higher is the elastic energy. High elastic energy leads to re-bounces and higher needs for the thusters
- For effective and robust energy absorption at different landing angles, the crushable should have the capability to deform also in direction orthogonal to the leg.

13.3 Baseline Design

13.3.1 ERC Spin and Ejection Device

The Spin and Ejection Device on the ERC will provide the impulse for the separation from the ERV. The actuation force is provided by 3 springs. The axes of the springs can be oriented with an angle with respect to the S/C velocity axis, in order to provide a certain spin, if this is deemed necessary. Few rpm of rotational speed can be easily obtained. The separation speed is about 0.25 m/s.

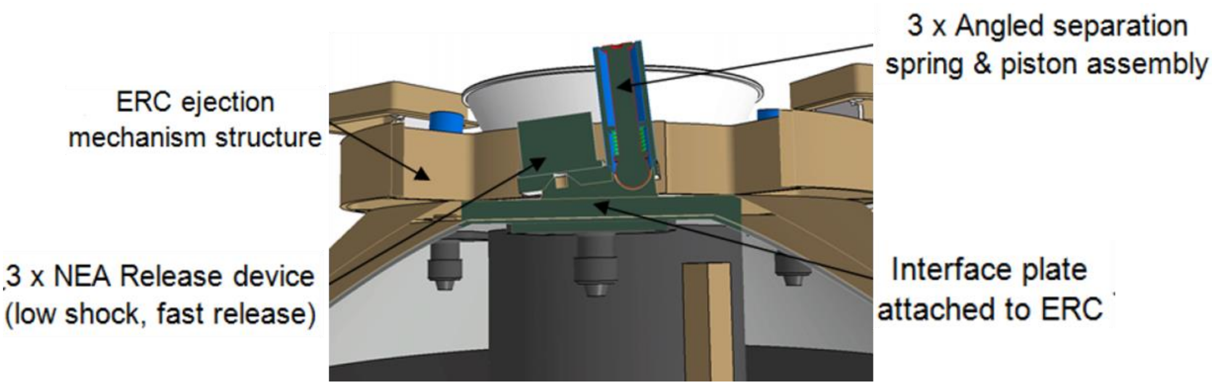

Figure 13-5: Conceptual solution for the ERC SED. The structure of the ERV which holds the ERC is located on the top, and the ERC hung under it

The first travel length of the ERC will occur inside the ERV, for about 0.5 m. Any impact between the ejected ERC and the internal walls of the ERV has to be avoided. Sufficient radial clearance shall be foreseen, according with the maximum lateral error speed. Assuming a lateral error speed of *10%* of the axial one, a clearance cone of semi-angle *arctan(0.1)= 5.7 deg* results. The clearance needed at the exit of the ERV is about *5 cm*.

The ERC and ERV interface points are held together by NEA release devices. The relevant flanges carry the loads exchanged from the ERV and ERC, especially the launch loads. It is noted that during launch and landing on Phobos, the ERC is held by the ERV from the top, therefore the static components of the launch loads tend to separate the flanges from each other.

The SED springs and NEA have to be accommodated on a ring structure, with sufficient internal clearance to allow the insertion of the sample container by the robotic arm.

Total mass of the SED is about 1 kg, of which only 0.2 kg remain on the ERC after separation.

The TRL for this mechanism is 3, since only a conceptual design exists.

A sufficient synchronization of the release devices is needed as well as the alignment of the spring actuation axis and equivalent forces, in order to have a speed direction as close as possible to the ERV axis, and to minimise the rotational speeds of the ERC along the other two orthogonal axis. Also the CoG of the ERV and ERC should be as much as possible aligned to the S/C velocity axis.

13.3.2 Sample Container

The sample container stores the sampled soil collected by the sampling tool, while attached to the tip of the robotic arm, and is then transferred and secured into the ERC.

Its design strictly depends on the solution of the sampling tool. With a sampling concept based on rotating bristles (brushes), the sample container shall be provided with a central aperture which allows capturing the particles of soil lifted up by the bristles. The containment volume shall be enough to carry the desired amount of sample (*100 g*). It needs a mechanism (doors) to close the aperture and hold the particles once the sampling operations are finished. Suitable instruments shall provide a measurement of the quantity of soil collected. Care shall be taken in the design to ensure that trapped particles cannot jam the mechanisms.

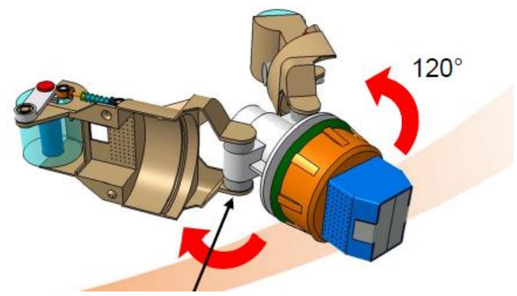

Figure 13-6: Conceptual representation of the sample container while attached to the sampling device (Airbus DS concept from the Phootprint study). In this example, the bristles open to allow the container to be inserted into the ERC

Once placed on the ERC, the sample container needs a mechanism to firmly hold it in place, throughout the whole environmental conditions of return and re-entry to Earth.

The placement of the sample container shall be done only by the robotic arm, therefore other actuated mechanisms should be avoided. A kind of threaded interface will need the arm to rotate around the container axis. A bayonet mechanism, as that depicted on the figure below, probably needs a higher insertion force by the robotic arm (around 40N).

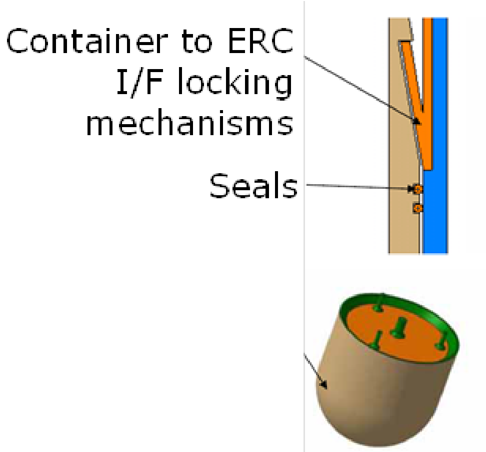

Figure 13-7: Conceptual representation of the holding interface for the sample container (Airbus DS concept from the Phootprint study). A bayonet-type mechanism is shown in this example

Appropriate sealing is needed. The level of sealing performance and reliability has to be consistent with the sample protection needs of the mission.

At the re-entry, the mechanism will undergo heavy thermal and mechanical loads, which also will characterise a lot of its design and testing requirements.

13.3.3 Landing Gear

Baseline design foresees 4 legs with crushable material. Depending on the required deceleration and legs angle the crushing force should be in the range 500-2000 N. No viscous damper or elastic element has been considered. This means that the S/C might be misaligned with respect to the soil after landing, due to lateral speed at touch-down or slope of the soil (unknown).

A solution with only three legs has not been analysed in the frame of this study. The 4 legs solution was assumed to be more robust, considering the uncertainties about the Phobos soil shape and mechanical properties.

The legs footprint accommodation within the fairing of the launch vehicle (Proton) has been taken into account. Hence an available diameter of 4.2 m was considered. The length of the legs should be enough to allow a sufficient clearance under the S/C after landing, considering both the presence of exposed rocks, possible bumpers on the S/C, and height reduction due to crushed material.

The resulting configuration of the legs produces a quite high ratio between the height of the CoG and the legs footprint radius, if we compare it with past studies. As discussed in a previous section, this reduces the stability margins, which has to be compensated by a sufficient hold-down thrust force and proper design of crushable material properties and legs elastic behaviour.

13.3.4 High Gain Antenna Pointing Mechanism (HGA APM) and Hold Down and Release Mechanism (HDRM)

For the pointing of the HGA, and its pointing in azimuth and elevation, the baseline is the Antenna Pointing Assembly designed by KDA for Bepi Colombo (Figure 13-8). Two hold down and release devices are foreseen.

For the APM to withstand the landing acceleration, the accommodation and attachment of the antenna shall consider to minimise the momentum arm from the antenna CoG and the bearings of the APM. Furthermore when landing, the antenna shall be placed along the S/C panel, like in the stowed configuration, therefore minimising the arm of the vertical load passing through the CoG of the antenna. In this way, resettable HDRM can be avoided in the baseline solution. Anyway a more accurate analysis of the touch-down should be done to confirm that the acceleration on other direction, or coming from bouncing and other rotations, will not induce critical loads on the APM bearings.

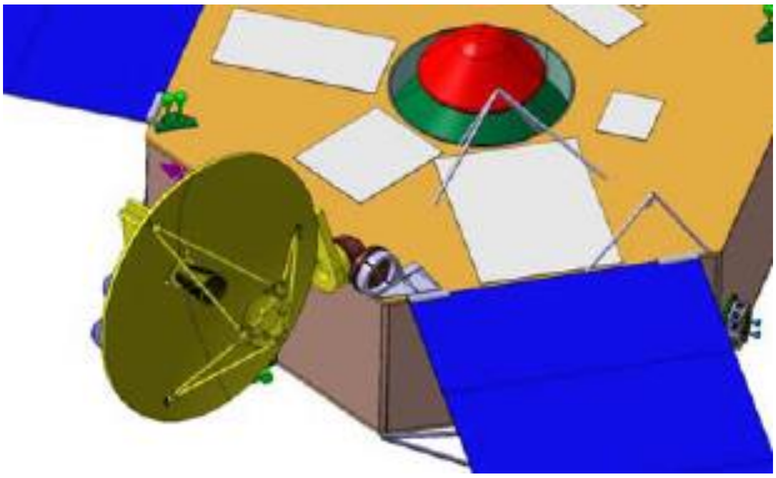

Figure 13-8: Bepi Colombo high gain antenna pointing mechanism and HDRM

13.3.5 Solar Array Deployment Hinges and Latches and Hold Down and Release Mechanism (HDRM)

For this study, the tracking of the sun is not required, therefore the SA will not require a drive mechanism.

Deployment hinges and latches are therefore used. Hereafter, the loads on the hinges, specifically the root hinges, are assessed during touch-down. An acceleration of 1 g is assumed. Hold down and Release Mechanisms, 4 per each wing, are foreseen to support the launch loads.

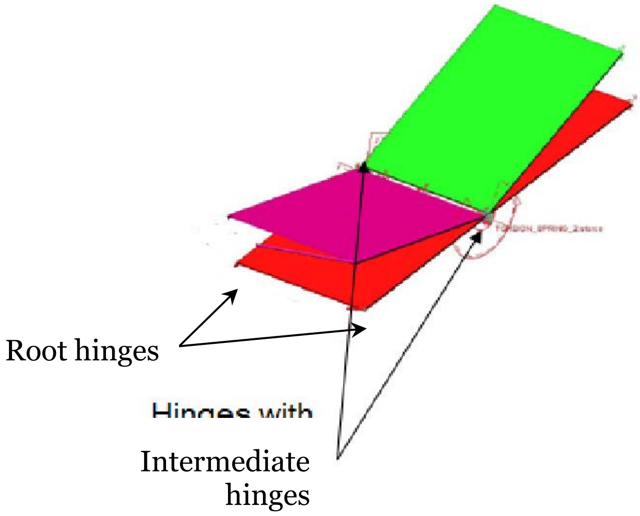

Figure 13-9: Schematic view of 2 foldable solar panels in one wing and hinges locations

The solar array area is assumed to be *11 m²*, is made up 6 wings and each wing has 2 foldable panels connected by 2 intermediate hinges. A surface mass density of *4 kg/m²* is assumed for the calculation of the loads. Each of the 6 wings has a mass of about *7.4 kg*, width *0.82 m*, and length of *2.4 m*. Since the load is seen as a step function, and the response of the hinges is assumed perfectly elastic, a maximum load factor of 2 is assumed. Every hinge line has 2 hinges. With these hypothesis, the load on each hinge is estimated in *75 N*, the torque from the bending moment on the wing (arm *1.2 m*), *85 Nm*.

The load levels are compatible with those allowable of standard hinges employed in many past missions.

Note that the number of wings eventually has been decreased from 6 to 5 wings (while maintaining the same area) leading to a reduced number of SA deployment mechanisms and HDRMs, and thus providing about 5 kg of spare mass which will compensate the mass increase of each panel.

13.4 List of Equipment

In the following tables, the list of the equipment and mass budget with margins are given. Note that the sampling tool is described in chapters 24 and 25.

Element 1	Lander			MASS [kg]			
Unit	Unit Name Click on button above to insert new unit	Part of custom subsystem	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
1	Landing gears		4	5.6	To be developed	20	26.9
2	Sampling tool		1	4.1	To be developed	20	4.9
3	APM+APME+HDRM		1	12.5	To be modified	10	13.8
4	SA deployment mech. + HDRM		6	5.0	Fully developed	5	31.5
5	Robotic arm		1	14.3	To be developed	20	17.1
-	Click on button below to insert new unit						
SUBSYSTEM TOTAL			5	83.3		13.1	94.2

Table 13-1: Lander mechanisms equipment list

Element 2	ERC				MASS [kg]			
Unit	Unit Name Click on button above to insert new unit	Part of custom subsystem	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
1	SED		1	1.0	To be developed	20	1.2
2	Sample container		1	1.2	To be developed	20	1.4
-	Click on button below to insert new unit						
SUBSYSTEM TOTAL			2	2.2		20.0	2.6

Table 13-2: ERC Mechanisms equipment list

13.5 Options

13.6 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Included in this table are:

- Technologies to be (further) developed
- Technologies available within European non-space sector(s)
- Technologies identified as coming from outside ESA member states.

Equipment and Text Reference	Technology	TRL	Technology from Non-Space Sectors	Additional Information
Spin and Ejection Device	Spring actuated ejection, hold down and release.	4	No	Similar application are already mature and with heritage for components like spring actuation and release mechanisms.
Sample container	Sample enclosure, monitoring, sealing.	3	Yes	Level of sealing requirement should be evaluated (coming from planetary protection req.). Some activity are currently developing breadboard models.
Landing Gears	Stability, load dissipation, crushable material, simulation and testing techniques.	3-4	Yes	-

Note that the sampling tool technology requirements are addressed in chapters 24 and 25.

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14 PROPULSION

14.1 Requirements and Design Drivers

The requirements relating to the chemical propulsion system of the Phobos Sample Return mission's Lander are to provide, for the various mission phases and for various vehicle combinations:

4. The required attitude control capabilities (Lander RCS thrusters to control the propulsion module during its burns and coasting phases)
5. The required velocity increments and RCS (Lander)
6. The required thrust levels for main manoeuvres of the Lander.

The propulsion system design drivers are the performance characteristics of the various thrusters and available / modifiable propellant tanks. Other design drivers considered were:

- Preferably European COTS component selection
- Single fault tolerant system design.

14.2 Assumptions and Trade-Offs

A mass comparison between a monopropellant and a bipropellant propulsion system was carried out, which turned out to be favourable for the bipropellant system. However, for a series of advantages listed in the system chapter a monopropellant system has been assumed as the baseline for the Lander's propulsion system.

The Isp of the hydrazine system is assumed at 220 s.

A margin policy described in RD[17] has been adapted. The following margins have therefore been considered:

- 2% on propellant residuals
- 5% on total ΔV
- 100 % for AOCS propellant

The following masses have been used in the calculations (including margins):

- Propulsion module dry mass is 674 kg
- Earth Return Vehicle + Earth Return Capsule mass is 529 kg
- Lander dry mass in including the propulsion system is 721 kg.

Additionally, the propellant amounts left in the propulsion module (Russian delivered) in between different manoeuvres, were determined by the Systems discipline and are described in chapter 6 System.

The propellant required for reaction and roll control (provided by the lander) of this module with varying propellant amounts and with all the other modules on top during and in between burns (Lander, Earth Return Vehicle and Earth Return Capsule) was determined by the propulsion discipline. Together with the provision of the major delta v's, the total propellant amount was determined.

The number of RCS thrusters on the Lander is derived from the AOCS requirements. 16 thrusters of 20 N have been baselined by the AOCS discipline (with an option of 24 thrusters for a better roll control authority during the Fregat-based PM manoeuvres, as described in the GNC part, to be further analysed in future phases).

14.3 Baseline Design

The baseline design involves a monopropellant propulsion system with 16 reaction control thrusters with 20 N thrust magnitude each. These thrusters are used for typical reaction control manoeuvres and for providing main ΔV to the Lander (and its payload). These thrusters also control the attitude of the whole stack (Propulsion Module, Lander, Ascent Vehicle and Earth Return Capsule) during burns and in between burns of the Propulsion Module.

The system operates in blow down mode.

14.4 List of Equipment

Table 14-1 lists the equipment and the associated masses of the propulsion system with the 16 thrusters configuration. Note: In the system level mass budget the 24 thrusters configuration is considered, however the resulting mass impact on the overall mass budget is negligible.

The total amount of propellant to be loaded is 519 kg.

-			MASS [kg]			
Unit Name Click on button above to **insert new unit**	Part of custom subsystem	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
20N thruster		16	0.4	Fully developed	5	6.4
Propellant tank		4	18.500	To be developed	20	88.8
Propellant filter		1	0.300	Fully developed	5	0.3
Latching valve		2	0.700	Fully developed	5	1.5
Pressure transducer		3	0.280	Fully developed	5	0.9
Fill and Drain valve / Vent valve (propellant)		1	0.070	Fully developed	5	0.1
Fill and Drain valve / Vent valve (pressurant)		4	0.070	Fully developed	5	0.3
Piping (incl fittings)		1	2.000	Fully developed	5	2.1
Stand-off		1	2.500	Fully developed	5	2.6
Mounting screws		1	2.000	Fully developed	5	2.1
Miscellaneous		1	0.250	Fully developed	5	0.3
Pressurant		1	0.700	Fully developed	5	0.7
Click on button below to insert new unit						
SUBSYSTEM TOTAL		**12**	**90.4**		**17.3**	**106.0**

Table 14-1: Equipment list and associated masses

14.4.1 Thrusters (20 N)

The 20 N thrusters use the storable propellant hydrazine (N_2H_4) and are designed for both long term steady state and pulse mode operation.

The thrusters operate over a wide pressure range and are thus ideal for blow down propulsion systems.

The combustion chambers and nozzles are manufactured from Haynes 25 (Co-Ni-Cr-W) alloy. The structure is also designed to serve as a heat barrier for protecting both the propellant valve and spacecraft structure. The thruster is available with either a straight or canted nozzle (see Figure 14-1 below).

An internally redundant catalyst bed heater and thermal insulation guarantees optimum start up conditions. In addition, the thruster is qualified for multiple cold starts.

Thruster valves are of the dual seat solenoid type, produced by Airbus Space Systems and also by external partners. The solenoids operate at a nominal valve voltage of 28 volts DC and are mechanically and electrically decoupled from each other. Thruster performance is identical for both Airbus and partner produced dual seat valves.

14.4.2 Thruster Power Consumption

The theoretical power consumption of the propulsion system is described below.

If the **Lander** is active, the following has been assumed:

- Assume maximum 4 thrusters on, each using 13 W (thrusters monostable valve): 4 x 13 W = 52 W.
- During landing operations (no main engine); assume maximum 8 thrusters on of 13 W each : 8 x 13 W = 104 W

Additionally catalyst bed pre heating is required prior to thruster firing. For this, each thruster requires 3 W (prior to the burn, not during the burn).

Figure 14-1: 20 N thrusters with canted nozzle

14.4.3 Custom Designed Propellant Tank

Initially off the shelf propellant tanks were baselined (Herschel Planck heritage). However it turned out that 4 of these tanks would still be too small to contain the required propellant. Therefore new developed tanks have been assumed based on extrapolation of existing tank masses, thus considering 20% subsystem maturity margin.

14.5 Options

Different mission options in different years were considered. The baseline mission starts with a launch in 2024 which resulted in a propellant loading of 519 kg, based on ΔVs corresponding to that launch window. The back up launch opportunity is in 2026. This option considers different velocity increments, but the dry masses of all systems are kept the same. This results however in a slightly different propellant loading of578 kg. The propulsion system architecture remains unchanged.

14.6 Technology Requirements

All components, at the exception of the tanks, are off the shelf.

However, delta qualification might be required for e.g. the thrusters since these have only flown in Earth orbit, high elliptical Earth orbit and to the EML2 point, but never to Mars. Formally, this delta qualification might bump, back the TRL of the thrusters to 7, but there is no reason to assume why problems would occur in the delta qualification process.

15 GNC

15.1 Requirements and Design Drivers

Given the responsibility sharing agreed between ESA and ROSCOSMOS, the only GNC subsystem analysed has been the one of the landing module.

The lander GNC subsystem is in charge of the measurement and control of the composite attitude and position from launcher separation up to the end of the Deimos-Phobos transfer burn. Once the propulsion module is jettisoned, the GNC subsystem will also have to ensure the appropriate attitude and position of the landing module during the Phobos close proximity operations, descent and landing, and surface operations (including thruster activations during surface operations to avoid taking off from the surface).

The most critical phase of the mission deals with the close proximity operations around Deimos and Phobos and with the descent and landing (D&L) at Phobos. The main driving requirements of the D&L phase are the ones concerning the satellite characterisation and terminal conditions:

- LM-120: landing accuracy on Phobos better than 50 m (3-sigma)
- LM-140: landing velocities at Phobos (vertical <100 cm/s ; horizontal < 15 cm/s)
- LM-180: landing phase shall end by a 20 m (TBC) free-fall without using thrusters. Altitude of the free-fall is a result of system level trade-off (landing gear sizing, stability, soil contamination, etc…)
- MI-120, MI-130: 3 fly-by trajectories for high resolution measurements at altitudes lower than 5 km (TBC) of potential landing sites
- MI-140: 20% accessibility of Phobos surface (including latitudes up to 60 deg)

During cruise, the main responsibility of the GNC subsystem would be to ensure a sun pointing mode, allowing the composite or the lander module to track the sun with an estimated accuracy of approximately 1°. There is no need for an Earth pointing mode since the steering mechanism of the high gain antenna is expected to orient it towards the Earth during the sun pointing mode.

At the Phobos and Deimos characterisation phase, the scientific observations of the Martian moons with the narrow and wide angle cameras (NAC and WAC) drive the pointing accuracy requirements.

The GNC subsystem should also ensure the right orientation of the composite during the firings of the propulsion module. The main engine of the propulsion module (Fregat derived) is expected to be gimballed and capable of actively controlling pitch and yaw during the firings. Therefore the GNC subsystem of the descent module shall only be in charge of compensating any roll perturbation generated during the firings (to be quantified)

15.2 Assumptions, Trade-Offs and Specific Analysis

In the frame of this CDF study no detailed GNC analysis has been performed. The following paragraphs aim to provide some information from the recent Phootprint studies (RD[20], RD[21], RD[22]), justify the methodology and delta-v budget used as inputs for this CDF study, and make preliminary assessments on the feasibility of the landing requirements (to complement the landing gear analyses).

15.2.1 Phobos Environment Modelling

Based on Phootprint analyses (RD[18]), for Phobos' distances higher than 30-50 km it is advisable to use a Mars-centred model, using as perturbations Mars non-sphericity effects (J2 and J3) rather than for Phobos non-sphericity. For low altitude phases it is possible to switch to a Phobos-centred model, with Mars as spherical third body, although Mars-J2 effect would still represent a relevant perturbation (RD[18]).

Another important factor to take into account is the error on the gravitational forces, which will be a key driver in navigation analysis. Although the uncertainties on the experienced gravitational forces tend to be compensated by the closed loop when the GNC subsystem is operating, they will contribute to increase the landing errors during the free-fall phase. Therefore a detailed error budgeting is needed in order to assess which are the level of acceleration uncertainties to be expected when being very close to Phobos surface.

As shown in RD[23], the local gravity at Phobos' surface can vary from 0.0019 m/s^2 to 0.0084 m/s^2 over the dimensions of the moon of 26.8 x 22.4 x 18.4 km. Assuming compliance with the requirement LM-120 (3-σ landing accuracy better than 50 m), and assuming a variation of 0.0084 - 0.0019 = 0.0065 m/s^2 over 18.4 km, the resulting local gravity uncertainty for 50 m would be in the order of 1.7E-5 m/s^2. This of course is a very preliminary figure.

According to RD[20], Phobos' μ and J2 are known with a 1-σ dispersion of ±0.05% and ±36% respectively. The dominant effect in the acceleration error (computed at 11.1 km altitude from Phobos) comes from the J2 error (7.2E-4 m/s^2) as shown in Table 15-1. Since the landing phase will be preceded by an extensive phase of precise orbit determination from Earth, it is expected to reduce the uncertainty on J2 by at least a factor of 10, resulting in a 1-σ dispersion of 7.2E-5 m/s^2.

	Mars μ [km3/s2]		Mars J2		Phobos μ [km3/s2]		Phobos J2	
	nom	1σ	nom	1σ	nom	1σ	nom	1σ
	4.284E+04	7.400E-05	1.960E-03	1.000E-06	7.114E-04	3.557E-07	1.170E-01	4.200E-02
acc [m/s2]	5.010E-01	8.655E-10	2.947E-03	1.503E-06	5.774E-03	2.887E-06	2.027E-03	**7.275E-04**

Table 15-1: Mars and Phobos uncertainties in gravity model

Mars' gravity field is much better known than Phobos', and therefore the resulting acceleration created by the uncertainties in Martian μ and Martian J2 can be completely neglected (8.6E-10 and 1.5E-06 m/s^2).

Finally, the modelling of Phobos' gravity field by means of spherical harmonics is a good approximation of reality for distances beyond 14 km (RD[20]) but close to the surface it can have up to 7% error (1σ) on acceleration in certain specific areas, although in most of the surface the error will be 1% (1-σ).

So based on the above reasoning, and adding a safety factor of 1.5, two types of acceleration error models at Phobos' surface can be considered:

- Nominal error (1-σ) : 1.5% of local gravity + (1.7E-5+7.2E-5) x 1.5 = 1.3E-4 m/s²
- Conservative error (1-σ) : 10% of local gravity + 1.3E-4 m/s²

To conclude with the environment modelling, a first order estimation of the gravity gradient experienced by the S/C is computed using the following formulation (RD[24])

$$T_g = \frac{3\mu}{2R^3}\left|I_{yy} - I_{zz}\right|\sin(2\theta)$$

The worst case scenario with an off-nadir angle (θ) of 45 degrees is considered, together with a conservative difference of moments of inertia equal to 100 kg x m². The resulting gravity torque due to Mars gravity is 3.19E-6 Nm. Although the applicability of the same formula for a small body as Phobos would need to be further investigated in future phases of the study, the estimated gravity gradient created by Phobos at distances of 100 and 10 km is estimated to 7.78E-08 Nm and 9.88E-06 Nm respectively.

15.2.2 Free-Fall Phase

Requirement LM-180 forces the landing sequence to end with a free-fall phase with no engine active, in order to reduce contamination on the surface. It is obvious that the higher the altitude of the start of the free-fall, the lower the contamination, but a precise estimation of the contamination as function of the initial altitude and thrusters orientation is out of the scope of this activity.

The purpose of this section is to assess how much the landing accuracy is degraded during the free-fall phase, as a result of the uncertainty on the gravitational forces and the initial velocity errors at the beginning of the free-fall phase. The analysis will be done for two different initial altitudes (20 and 60 m) which can be used as inputs for the system trade-off needed to select the baseline initial altitude. Note that if the use of hydrazine is confirmed (which is less an issue for soil contamination than bi-propellant), it is expected that free fall altitude close to 20m is more likely.

The free-fall phase will be simulated using the following simplification: the main force is Phobos' surface gravitational force (either 0.0084 or 0.0019 m/s²), to be perturbed with two type of acceleration errors as described in section 15.2.1: nominal and conservative. As suggested in RD[20], the velocity error (3σ) considered at the beginning of the free-fall phase is 10.5 cm/s (which seems to be a quite conservative assumption based on the thruster performance accuracy analysis shown later in section 15.2.6).

		Gravitational acc		Initial conditions			Touchdown conditions			
Surface g level	Error in g (3σ)	g_surface [m/s2]	g_error (3σ) [m/s2]	h0 [m]	vv0 (3σ) [m/s]	vh0 (3σ) [m/s]	time [s]	vv [m/s]	vh [m/s]	downrange [m]
Low	Nominal	0.0019	0.00048	60	-0.105	0.105	313	0.49	0.182	40
	Conservative	0.0019	0.00096	60	-0.105	0.105	313	0.49	0.318	57
High	Nominal	0.0084	0.00077	60	0.105	0.105	108	1.01	0.134	12
	Conservative	0.0084	0.00291	60	0.105	0.105	108	1.01	0.330	20

		Gravitational acc		Initial conditions			Touchdown conditions			
Surface g level	Error in g (3σ)	g_surface [m/s2]	g_error (3σ) [m/s2]	h0 [m]	vv0 (3σ) [m/s]	vh0 (3σ) [m/s]	time [s]	vv [m/s]	vh [m/s]	downrange [m]
Low	Nominal	0.0019	0.00048	60	0	0.105	251	0.48	0.159	30
	Conservative	0.0019	0.00096	60	0	0.105	251	0.48	0.263	40
High	Nominal	0.0084	0.00077	60	0	0.105	120	1.00	0.139	14
	Conservative	0.0084	0.00291	60	0	0.105	120	1.00	0.363	24

		Gravitational acc		Initial conditions			Touchdown conditions			
Surface g level	Error in g (3σ)	g_surface [m/s2]	g_error (3σ) [m/s2]	h0 [m]	vv0 (3σ) [m/s]	vh0 (3σ) [m/s]	time [s]	vv [m/s]	vh [m/s]	downrange [m]
Low	Nominal	0.0019	0.00048	60	-0.105	0	313	0.49	0.149	23
	Conservative	0.0019	0.00096	60	-0.105	0	313	0.49	0.300	47
High	Nominal	0.0084	0.00077	60	0.105	0	108	1.01	0.083	4
	Conservative	0.0084	0.00291	60	0.105	0	108	1.01	0.313	17

Table 15-2: 3σ Touchdown conditions for 60 m initial altitude and different initial velocity errors

For a 60 m free-fall phase, the vertical velocity at touchdown is 1.01 m/s assuming a high surface acceleration. Starting with null vertical velocity barely affects the vertical velocity at touchdown (1.00 m/s in the second set of data of Table 15-2). Therefore 60 m seems to be the maximum altitude at which we can start the free-fall phase if we want to fulfil LM-140: vertical velocity < 1 m/s.

On the other side, the key parameter for the downrange error is the time to touchdown which is driven by the surface acceleration level (low level makes the free-fall phase last longer). From the 57 m maximum downrange, only 10 m are caused by the initial lateral velocity of 0.105 m/s (this result is obtained comparing downrange of the first and third set of results of Table 15-2). When starting the descent at null vertical velocity, the time is reduced to 251 s (in comparison to the 313s when the initial vertical velocity is upwards). Therefore it would be possible to increase the initial downwards vertical velocity (0.88 m/s) until reaching the limit of vertical velocity at touchdown (1 m/s). Taking into account the ±0.105 m/s realisation error, it is advisable to target 0.775 m/s as initial vertical velocity. These optimised initial conditions for 60 m altitude are shown in Table 15-3.

		Gravitational acc		Initial conditions			Touchdown conditions			
Surface g level	Error in g (3σ)	g_surface [m/s2]	g_error (3σ) [m/s2]	h0 [m]	vv0 (3σ) [m/s]	vh0 (3σ) [m/s]	time [s]	vv [m/s]	vh [m/s]	downrange [m]
Low	Nominal	0.0019	0.00048	60	0.775	0.105	71	0.91	0.110	8
	Conservative	0.0019	0.00096	60	0.775	0.105	71	0.91	0.125	8
High	Nominal	0.0084	0.00077	60	0.105	0.105	108	1.01	0.134	12
	Conservative	0.0084	0.00291	60	0.105	0.105	108	1.01	0.330	20

Table 15-3: 3σ Touchdown conditions for 60 m initial altitude and optimised initial vertical velocity

Following a similar strategy for the case of initial altitude 20 m, the results are presented in Table 15-4.

		Gravitational acc		Initial conditions			Touchdown conditions			
Surface g level	Error in g (3σ)	g_surface [m/s2]	g_error (3σ) [m/s2]	h0 [m]	vv0 (3σ) [m/s]	vh0 (3σ) [m/s]	time [s]	vv [m/s]	vh [m/s]	downrange [m]
Low	Nominal	0.0019	0.00048	20	0.855	0.105	23	0.90	0.106	2
	Conservative	0.0019	0.00096	20	0.855	0.105	23	0.90	0.107	2
High	Nominal	0.0084	0.00077	20	0.105	0.105	58	0.59	0.114	6
	Conservative	0.0084	0.00291	20	0.105	0.105	58	0.59	0.198	8

Table 15-4: 3σ Touchdown conditions for 20 m initial altitude and optimised initial vertical velocity

Summing up, it can be concluded that even with the conservative case of acceleration errors the degradation of landing accuracy during the free-fall phase will be below 8 m in 3σ. In a similar way the increase in lateral velocity error will be in the order of 9.3 cm/s (19.8 cm/s of horizontal velocity at touchdown compared to the initial vh0 of 10.5 cm/s) or 0.9 cm/s depending on the error model. Given the fact that the requirement LM-140 imposes a maximum lateral velocity at landing < 15 cm/s, the initial assumption of 10.5 cm/s of 3σ velocity error at the beginning of the free-fall (suggested by RD[20]) may need to be revisited in order to fulfil LM-140. In reality with the proposed RCS configuration presented in section 15.2.6 the velocity accuracy which can be obtained taking only into account the error due to the MIB of the thrusters is in the order of 0.03 cm/s. Therefore it seems likely that the velocity error at the beginning of the free-fall taking into account the complete GNC errors chain will still be much lower than 10.5 cm/s.

15.2.3 Preliminary Landing Stability Assessment

A simplified assessment of the dynamic landing stability has been performed based on angular momentum considerations and additional simplifications detailed hereafter. These analyses complement those presented in the landing gear section (chapter 13.2.1), and feature very simple assumptions (no rebounce, no model of the leg dampers, …), so conclusions must be considered with care, more detailed analyses will be needed in future phases.

The first assumption is a two-dimensional movement, as presented in Figure 15-1. Additionally it is assumed that at the moment of impact, the leg will encounter some kind of obstacle which would prevent it from sliding and the whole vehicle will start a rotational movement around the impact point (point A in Figure 15-1). The initial angular rate $\dot{\theta}$ after touch-down will be computed applying the conservation of angular momentum between the two instants prior and after the impact:

$$\dot{\theta} = \dot{\theta}_0 + \frac{m \cdot v_h \cdot \sqrt{l^2 + h_{CoG}^2} \cdot \sin(\theta)}{I_{XX@CoG} + m \cdot (l^2 + h_{CoG}^2)}$$

where $\dot{\theta}_0$ is the angular rate prior to impact, m is the mass of the lander composite, v_h the horizontal velocity prior to impact. The denominator represents the moment of inertia around the impact point A. In order to make an assessment on which are the main drivers on the $\dot{\theta}$ computation let us consider the input data provided in Figure 15-5, in addition to a V_h equal to 0.15 m/s , α_o = 10 deg and initial angular rate of $\dot{\theta}_0$ of 0.1 deg/s (conservative compared to the 0.02 deg/s estimated in section 15.2.6). The result is

$$\dot{\theta} = \dot{\theta}_0 + 2.34\ deg/s$$

which clearly shows the small influence of $\dot{\theta}_0$ compared to the contribution of the lateral velocity (one order of magnitude less).

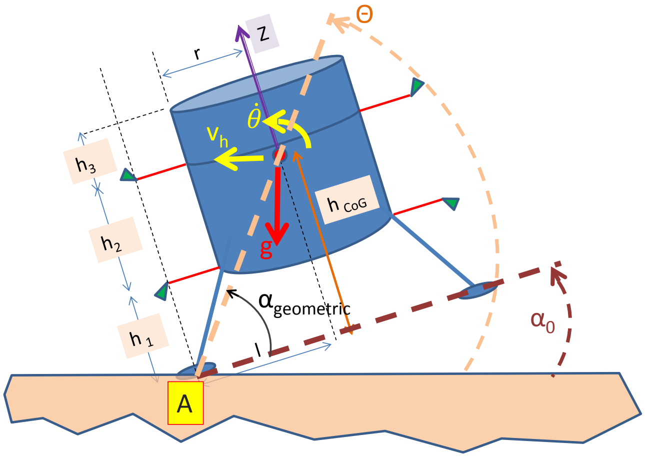


Figure 15-1: Preliminary landing stability assessment

During this rotational movement the gravitational acceleration of Phobos creates an angular deceleration which tends to counteract the initial angular velocity. This angular acceleration depends on θ and it is computed using the following formula:

$$\ddot{\theta} = -\frac{g \cdot m \cdot \sqrt{l^2 + h_{CoG}^2} \cdot \cos(\theta)}{I_{XX@CoG} + m \cdot (l^2 + h_{CoG}^2)}$$

At the instant of touchdown the angular acceleration $\ddot{\theta}$ can vary between 3.6E-02 to 9.9E-02 deg/s² depending on which surface gravitational acceleration is considered (1.9E-03 or 8.4E-03 m/s²).

The above equations have been integrated over time for different initial angular offset with respect to the surface (α_o). Note that in all the analyses it is assumed that the surface is a geodetic surface, meaning that it is perpendicular to the gravitational acceleration. The roll-over condition occurs when the angle θ exceeds 90 degrees. Figure 15-2 shows the evolution of the angle θ versus time for different surface gravitational values and different initial lateral velocities. For a g-level of 0.0084 m/s² and a V_h equal to 0.15 m/s, the maximum initial tilt with respect to the geodetic surface (α_0) is 4 degrees. For α_0 = 5 degrees, θ reaches 90 deg approximately 43 seconds after touchdown, meaning that the vehicle will roll-over. The g-level is a very sensitive parameter for this type of analysis: if the landing was to occur at a slightly lower gravity zone (g = 0.007 m/s²) the maximum α_0 would be reduced to barely 2 degrees (see Figure 15-2). In the same figure, the plots in the bottom show the influence of the touch-down lateral velocity.

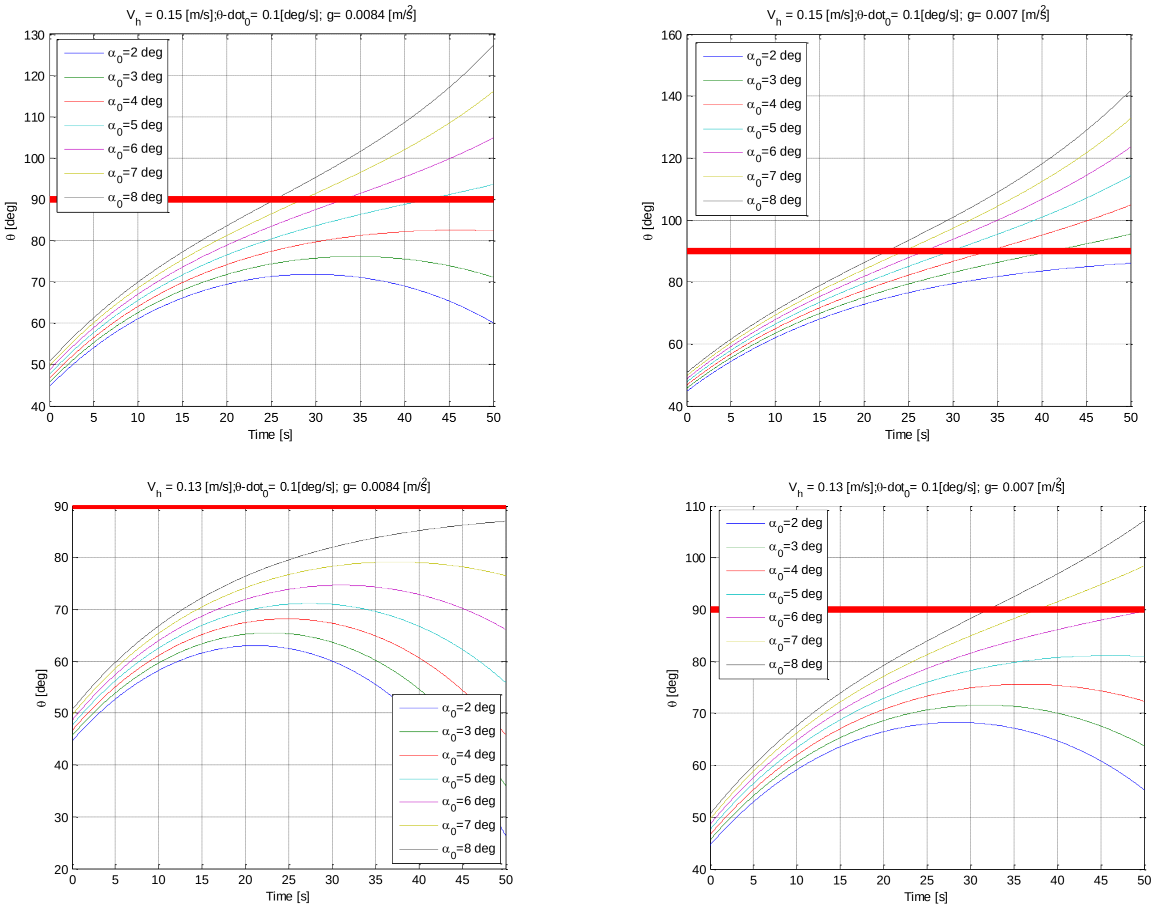


Figure 15-2: Rotation angle θ after touchdown. High surface gravitational level (left) and low surface gravitational level (right). Touchdown lateral speed 15 cm/s (top) and 13 cm/s (bottom)

Assuming that the free-fall phase can last 58 or 108 seconds (see Table 15-3 and Table 15-4) depending on the initial altitude (20 or 60 m), an initial angular rate of 0.02 deg/s as estimated in section 15.2.6 would imply an angular excursion of 1.2 or 2.2 deg respectively. In case we would relax the attitude requirement to 0.1 deg/s the resulting angular excursion at landing would be 5.8 or 10.8 degrees, which would not be acceptable based on the preliminary results of Figure 15-2. This could be corrected by operating the 12Nms reaction wheels which are the baseline attitude actuators for the close proximity operations. The maximum torque of these reaction wheels is around 0.15 Nm, which results in an angular acceleration of 0.01 deg/s² assuming an inertia of 711 kgm². With this acceleration the time to cancel the initial angular rate of 0.1 deg/s would only be 10 seconds having enough time to bring back the lander to the desired attitude prior to landing (free-fall phase duration between 58 and 108 seconds). However the use of reaction wheels during landing has not been baselined for the time being.

A parametric analysis has been performed for different conditions of gravitational acceleration at surface, touchdown lateral velocity, CoG location, distance from longitudinal axis to the legs (l) and moment of inertia at the CoG. The angular rate at touchdown $\dot{\theta}_0$ has been assumed to be 0.1 deg/s for all the cases, although its effect on the final results is almost negligible.

g_surface [m/s2]	vh [m/s]	h_CoG [m]	l [m]	Ixx@CoG [kg*m2]	Max initial α_0 [deg]
0.0084	0.15	1.83	1.993	711	**4**
0.0070	0.15	1.83	1.993	711	**2**
0.0065	0.15	1.83	1.993	711	**1**
0.0019	0.077	1.83	1.993	711	**1**
0.0084	0.15	1.83	1.993	711	**4**
0.0084	0.13	1.83	1.993	711	**8**
0.0084	0.10	1.83	1.993	711	**14**
0.0084	0.15	1.83	1.993	711	**4**
0.0084	0.15	1.73	1.993	711	**5**
0.0084	0.15	1.63	1.993	711	**7**
0.0084	0.15	1.83	1.993	711	**4**
0.0084	0.15	1.83	2.100	711	**6**
0.0084	0.15	1.83	2.300	711	**9**
0.0084	0.15	1.83	1.993	711	**4**
0.0084	0.15	1.83	2.100	1200	**5**
0.0084	0.15	1.83	2.300	2000	**6**

Table 15-5: Maximum initial angular offset α_0 at touchdown to avoid roll-over

The most important conclusion to be derived from the above results is that landing at low gravity regions (g = 0.0019 m/s^2) would require thruster activations at landing to avoid roll-over in case the lateral velocity is 0.15 m/s. To avoid thruster activations, the requirement of maximum lateral velocity needs to be increased to 0.077 m/s.

Results presented in Table 15-5 can be used to provide a preliminary requirement on the attitude orientation at touchdown to avoid roll-over and from there derive the necessary requirements for the GNC at the beginning of free-fall phase and during the operations of the reaction wheels during free-fall.

From the analysis above it can be seen how the attitude estimation relative to Phobos gravitational force is a key parameter for a safe landing. An attitude computation based on classic gyro-stellar estimation, coupled with a priori knowledge of the attitude of Phobos (ephemeris), can for sure provide attitude knowledge below 10 degrees (RD[21]) but further investigations may be needed to design a more accurate attitude estimation system for the last part of the descent.

15.2.4 Phobos Close Proximity Operations

The main characteristic of the Phobos-Mars dynamic environment is that it is not possible to consider classical Keplerian motion around Phobos due to the mass-distance relation between Mars and its moon. (RD[18]). Phobos' sphere of influence lies at approximately 7 km, which means that is actually contained entirely within the moon (Phobos' mean equatorial radius is 11.1 km). Fortunately, there does exist a trajectory type that combines low altitudes, stability and ease-of-maintenance: the so-called quasi-satellite orbit (QSO).

Such kinds of orbits are located beyond the Lagrange points, as seen from the rotating synodic reference frame. The shape of the QSO is given as the width of the orbit projected in the Phobos' orbital plane (similar to semi-major and semi-minor axis, where the semi-major axis is always oriented in the direction to Mars). Figure 15-3 shows a sample QSO of 28x44 km, as presented in RD[18].

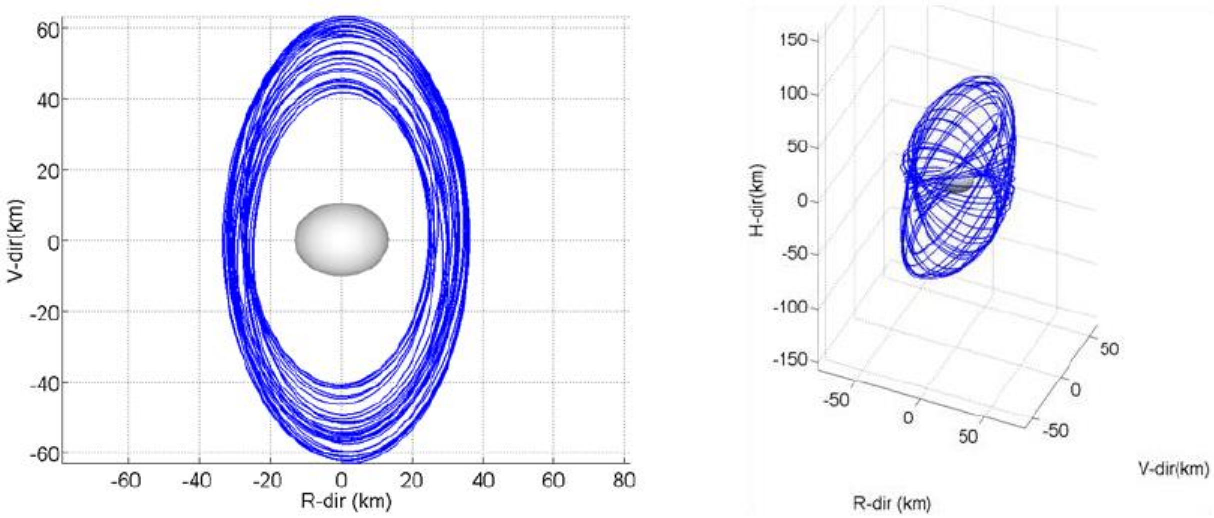


Figure 15-3: QSO width 28x44 km, 60 deg inclination (from RD[18])

These types of orbits are the result of an optimisation process where the initial velocity is optimised such that the resulting trajectory is stable over a period of time in which no other type of control is applied. This design approach allows obtaining a periodically controlled trajectory and an estimation of the dv budget.

Only certain combinations of the two dimensions of the QSO allow stable trajectories (those trajectories which require delta-v corrections at very short intervals are not feasible from an operational point of view). RD[18] and RD[21] consider three types of orbit: 50x100 km, 40x90 km and 28x44 km. The smaller the size of the QSO is, the higher is the delta-v budget for orbit maintenance. In a similar way, QSO orbits are more stable when they are within the orbit plane but orbits with high inclinations (up to 60 degree) are also feasible at a higher maintenance delta-v costs and more frequent manoeuvres. For example, for a QSO 40x90 km 0 deg inclination the dv budget is 0.011 m/s/week with a manoeuvre execution every four days, while for the same QSO but with a 60 deg inclination the delta-v increases to 0.373 m/s/week and a manoeuvre is required every day to ensure stability of the QSO (for more details see RD[18]).

To better characterise Phobos' surface at high latitudes there is the possibility to perform fly-bys departing from QSO. Given the manoeuvre execution errors the minimum altitude for a safe fly-by is preferred to be 5 km. Higher initial QSO inclinations allow lower delta-v cost for deorbiting when a high latitude fly-by is selected. Target fly-by latitudes of 60 degrees can be obtained even from 0 deg QSO with small increases in the departure dv, although in any case the deorbiting manoeuvre can always be kept below 5 m/s approximately. After the fly-by is executed, the S/C will start to drift-away from Phobos. Therefore two additional manoeuvres (phasing and reacquisition) will be needed, whose total delta-v cost depend on the total time from QSO departure to reacquisition. To give an order of magnitude, these two manoeuvres can be estimated on 10 m/s approximately.

Due to the time required to process the measurements, estimate the state and compute the manoeuvres an operational delay of approximately 8 hours should be considered between the last useful measurement and a burn execution. RD[18] provides results of a guidance and navigation montecarlo analysis for a QSO with 28x44 km and 60 deg inclination. This is the most demanding QSO in terms of manoeuvre frequency (one per day). Including propulsion execution errors, 8 hours operational delay and 1 manoeuvre per day, the montecarlo campaign shows that position and velocity error can be kept in the order of 100 m and 30 mm/s (1-sigma). This result proves that QSO computation strategy is feasible from G&N point of view. The increase in delta-v in comparison to the deterministic delta-v can also be estimated through this type of montecarlo analysis.

During the close proximity operations the S/C will need to be oriented towards Phobos' surface continuously in order to allow scientific operations. A sensor suite based on 2 star trackers and 2 sun sensors (together with two IMU's) provide suitable attitude determination performance and sensing. A set of reaction wheels will be used to ensure the necessary pointing accuracy. Based on the estimation of the gravity gradient torque presented in section 15.2.1, the accumulated angular momentum over one Phobos' orbit around Mars would be 0.09 Nms. This means that with the selected 12Nms reaction wheels, the wheel off-load manoeuvres should take place only once every 140 orbits around Mars (or equivalently once every 43 days approximately). Even taking 100% margin on this preliminary estimations for the frequency of off-loading manoeuvres it is safe to assume that these manoeuvres will be much less frequent than the QSO maintenance manoeuvres.

15.2.5 Phobos Descent and Landing

The driver requirement for selecting the appropriate descent and landing strategy is the LM-120 landing accuracy requirement of 50 m (3-sigma).

Two main options can be envisaged for the descent:

- Direct descent from QSO
- Descent to an intermediate body-fixed hovering

The direct descent from QSO requires an initial delta-v manoeuvre of 5-10 m/s, which assuming a 2% execution error implies a velocity error of 100-200 mm/s (3-sigma). Since the QSO knowledge velocity error including the 8 hours operational delay is around 50 mm/s it is clear that the execution errors become the driver dispersions for the direct descent scenario. Based on montecarlo simulations presented in RD[21], the estimated dispersion at Phobos surface using this direct descent strategy would be 2 km (3-sigma), which clearly exceeds the landing accuracy requirements.

The only possibility to increase the accuracy is to include a trajectory correction manoeuvre (TCM) somewhere along the 1-6 hours duration of the flight from QSO de-orbitation to Phobos surface. This would imply the use of additional on-board sensor data capable to provide velocity knowledge error better than the execution errors, together with an autonomous guidance. With the use of Doppler measurements together with the Mars Limb measurement using the science camera, montecarlo simulations (RD[21]) have shown that with a TCM of approximately 4 m/s performed 10 minutes before the target point, the final dispersions (relative to the landing site) at the start of the final forced descent can be reduced to approximately 100 m (3sigma). This accuracy is still not compliant to the landing accuracy requirement (50 m) but the inclusion of 2 TCMs instead of 1 could further improve the results.

Another strategy to perform the descent phase envisages an autonomous body-fixed hovering phase at low altitude to allow the acquisition of surface images to be sent to Earth, the selection of the landing site and the on-ground generation of the final descent trajectory. The hovering phase is supposed to make use of a wide angle camera (WAC) feature tracking and altimeter measurements to perform an autonomous anchoring to the nadir point.

RD[18] provides a detailed parametric analysis of the hovering phase for different altitudes and guidance strategy. Based on these results the preferred guidance scheme is the one based on velocity inversion at constant relative time intervals (i.e. 120 seconds, to take into account possible operational implementation limits due to on-board navigation estimation). The typical burn size is in the order of 0.5 m/s for hovering at 0 deg latitude and 0.5-1.5 m/s for the 45 deg inclination case. This imposes a requirement on the minimum controllability velocities to be provided by the RCS of the lander (see section 15.2.6). The overall delta-v cost for maintaining 1 hour of hovering at 1km altitude and 45 deg inclination is estimated in 15 m/s, decreasing to 8 m/s for hovering altitudes of 6 km. The results from a preliminary montecarlo campaign including burn execution errors and G&N has shown that the lateral drift can be kept below 50 m for a 2.5h hovering duration.

Finally, the landing phase will start from the hovering point or from the point targeted at QSO de-orbitation. For the case of direct descent, the final descent will need to be fully autonomous with no possible guidance update computed on-ground. In addition the direct descent strategy does not provide a go/no go decision to be made on-ground, increasing the risk and landing accuracy errors. On the contrary, the use of the body-fixed hovering would allow the on-ground computation of the guidance position/velocity profile based on latest navigation information. This strategy needs however to be further analysed in order to verify its feasibility, given the short time available for the loop with the ground during hovering (around 2.5h, due to the fact that the landing site must be kept illuminated during the “ vision based anchoring”, and the descent).

The proposed guidance scheme would be based on an Apollo-like guidance law that ensures arrival with zero velocity (RD[22]), controlling relative altitude and relative velocity. Based on preliminary montecarlo simulations presented in RD[22] this type of guidance would fulfil landing requirements, providing landing lateral velocities below 5 cm/s, lateral displacement below 18 meters and attitude errors below 5 degrees.

15.2.6 RCS Thruster Configuration and Sizing

The thrusters configuration needs to provide pure torque and pure force manoeuvres in order to control both position and velocity of the spacecraft without inducing any coupling with the attitude control.

The most optimised configuration shown later in this section is based on 2 x 8 thrusters (8 sets of two thrusters each to have full redundancy).

A thrust level of 20N for each thruster will be shown to be a good compromise between a high level thrust needed to reduce burn durations (and therefore gravity losses) and a low thrust level beneficial for precise attitude and orbit control during the descent phase.

The only reason for selecting higher thrust levels for the RCS thrusters would be the need to compensate disturbance torques during propulsion module main engine firings. Nevertheless, it is assumed that the Fregat-derived propulsion module is capable to control pitch and yaw channels through the thrust vector control (TVC) system of the engine.

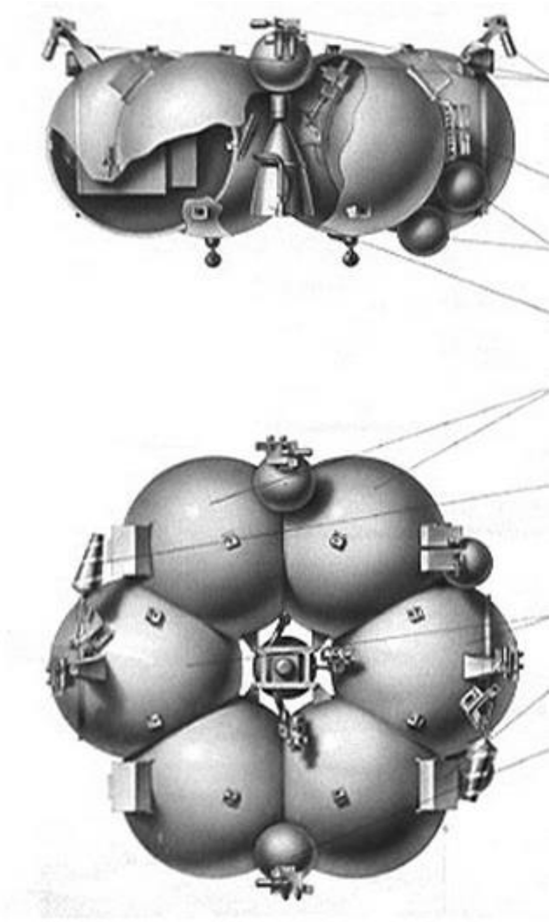


Figure 15-4: Fregat upper stage RD[25]

Figure 15-4 shows a sketch of the Fregat upper stage. The attitude control and main engine ignition thrusters (SOiZ) are composed of 4 clusters of 3 x 50 N thrusters each. Two of these thrusters are tilted approximately 45 deg while the third one is placed horizontally for roll control. The diameter and altitude of the Fregat are assumed to be 3350 and 1500 mm respectively. The pitch, yaw and roll torque which can be generated by the SOiZ is estimated to be [154, 154, 168] Nm. Given the fact that the main engine thrust is 20 kN, and assuming a CoG offset of 3 cm, the perturbing torque generated during the main engine firing would be in the order of 600 Nm, well beyond the torque capability of the SOiZ thrusters. This clearly demonstrates that the assumption of a TVC fully capable of controlling pitch and yaw during main engine firings is reasonable. The same capability is expected to be present in the propulsion module of this mission. There is no information available regarding the roll torque disturbance generated when firing the main engine of the Fregat, and therefore it is difficult to assess how much of the 168 Nm roll capacity is needed to attenuate the roll motion during main engine firings and how much is needed for attitude control of the upper stage or for spinning the payload before injection. In order to be conservative, a second configuration has been analysed including additional 8 roll thrusters to the original 8x2 thrusters. The results will be presented in the following paragraphs.

Figure 15-5 provides a sketch with the input data assumed to perform the GNC analysis.

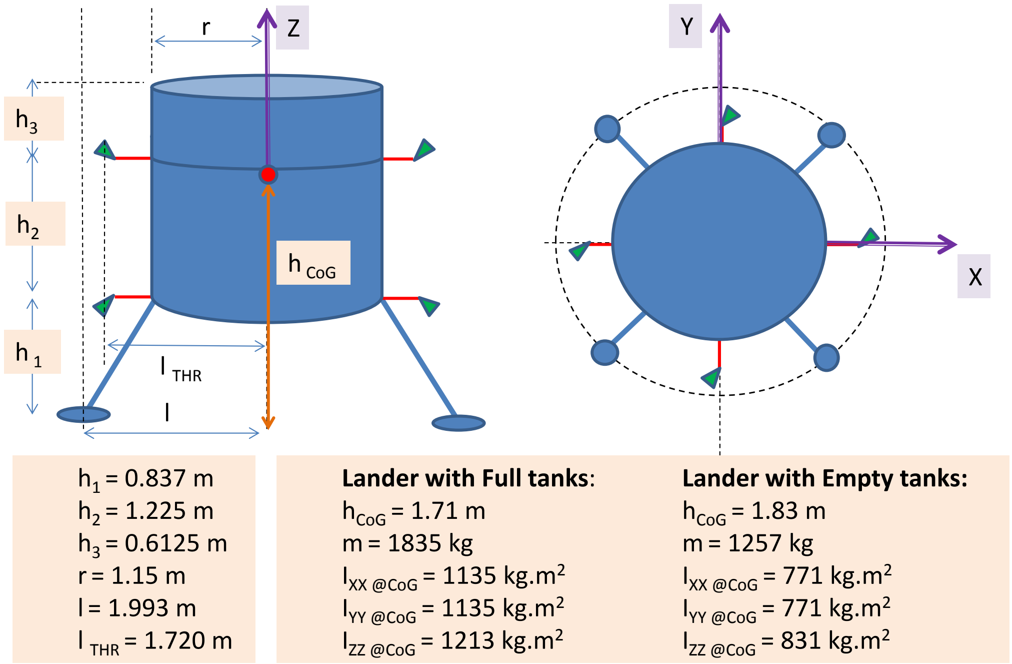


Figure 15-5: GNC simplified dimensions and MCI of Lander+ERV

	Position [m]			Unit vector Force		
Thruster #	x	y	z	x	y	z
1	1.720	0.000	0.837	-0.6830	-0.1830	0.7071
2	0.000	1.720	0.837	-0.1830	-0.6830	0.7071
3	-1.720	0.000	0.837	0.6830	0.1830	0.7071
4	0.000	-1.720	0.837	0.1830	0.6830	0.7071
5	1.720	0.000	2.062	-0.6830	-0.1830	-0.7071
6	0.000	1.720	2.062	-0.1830	-0.6830	-0.7071
7	-1.720	0.000	2.062	0.6830	0.1830	-0.7071
8	0.000	-1.720	2.062	0.1830	0.6830	-0.7071

Table 15-6: Position and orientation of the RCS thrusters for 15 deg azimuth

Figure 15-6 shows the orientation of the 8 reaction control thrusters for different configurations. Each arrow represents a hydrazine 20N thruster. The configuration is composed in reality of 16 thrusters for redundancy purposes, but only 8 are displayed in the plot. All the thrusters are tilted 45 deg with respect to the vertical while the azimuth angle has been varied from 0, 15, 22.5 and 45 deg.

Table 15-7 shows what is the maximum pure force and pure torque for each RCS configuration. The columns in the left represent the thrust level (from 0 to 1) requested to each thruster. The results have been obtained using a simplex algorithm implemented in Matlab/Simulink (RD[26]), which optimises the thruster firing to provide the commanded torque and forces while minimising the total firing energy of all the thrusters.

The configurations with azimuth angles of 15 or 22.5 deg are the preferred options, since they provide pure torque and force in all directions. When orienting the thrusters at an azimuth of 0 deg no roll-only (z-torque) authority is available, while when using an azimuth of 45 deg no pure force in x and y axis is possible.

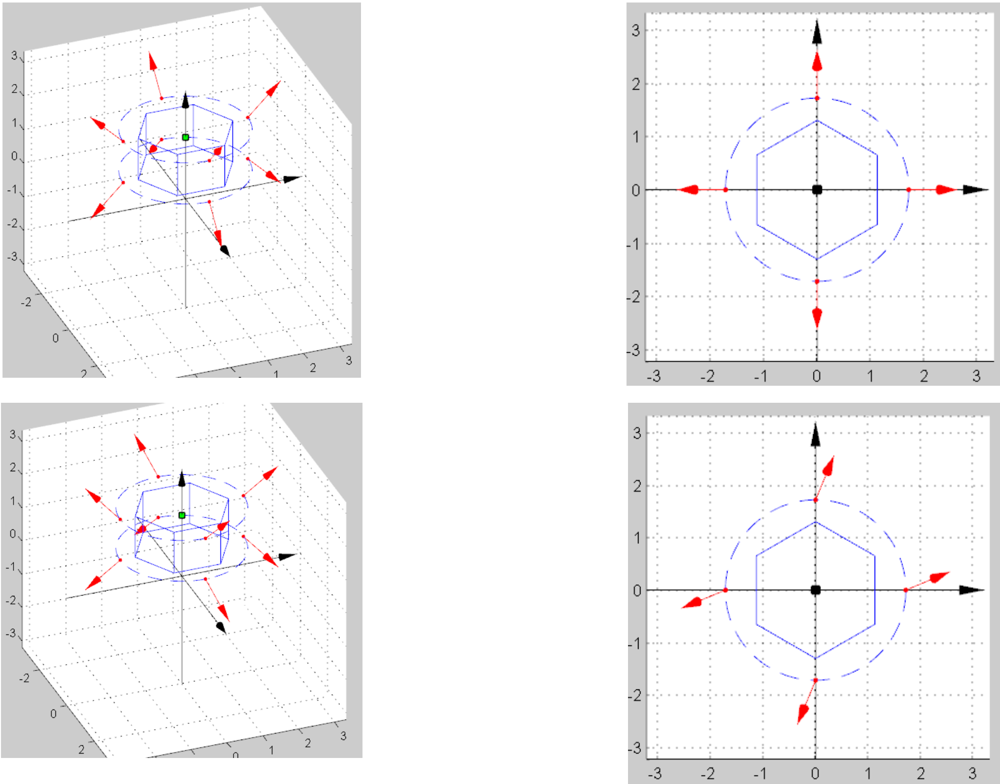


Figure 15-6: Configurations with 8 RCS tilted 45 deg wrt vertical. Azimuth angle equal to 0 deg (top) and 22.5 deg (bottom)

Maximum pure force and pure torque. Tilted 45 deg. Azimuth 0 degrees													
Thrust level (0 to 1) for each 22N Thruster								Fx	Fy	Fz	Tx	Ty	Tz
1	2	3	4	5	6	7	8	[N]	[N]	[N]	[N*m]	[N*m]	[N*m]
0.00	0.00	1.00	0.00	0.00	0.26	0.49	0.26	**23.2**	0.0	0.0	0.0	0.0	0.0
0.00	0.00	0.00	1.00	0.00	0.26	0.00	0.74	0.0	**23.2**	0.0	0.0	0.0	0.0
1.00	1.00	1.00	1.00	0.00	0.00	0.00	0.00	0.0	0.0	**62.2**	0.0	0.0	0.0
0.00	1.00	0.00	0.00	0.00	0.00	0.00	1.00	0.0	0.0	0.0	**34.5**	0.0	0.0
0.0	0.0	1.0	0.0	1.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	**34.5**	0.0
0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	**0.0**
1.00	0.00	0.00	0.00	0.74	0.00	0.26	0.00	**-23.2**	0.0	0.0	0.0	0.0	0.0
0.00	1.00	0.00	0.00	0.26	0.49	0.26	0.00	0.0	**-23.2**	0.0	0.0	0.0	0.0
0.00	0.00	0.00	0.00	1.00	1.00	1.00	1.00	0.0	0.0	**-62.2**	0.0	0.0	0.0
0.00	0.00	0.00	1.00	0.00	1.00	0.00	0.00	0.0	0.0	0.0	**-34.5**	0.0	0.0
1.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0	0.0	0.0	0.0	0.0	**-34.5**	0.0
0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	**0.0**

Maximum pure force and pure torque. Tilted 45 deg. Azimuth 15 degrees													
Thrust level (0 to 1) for each 22N Thruster								Fx	Fy	Fz	Tx	Ty	Tz
1	2	3	4	5	6	7	8	[N]	[N]	[N]	[N*m]	[N*m]	[N*m]
0.00	0.35	1.00	0.18	0.00	0.62	0.53	0.38	**21.3**	0.0	0.0	0.0	0.0	0.0
0.35	0.00	0.18	1.00	0.62	0.00	0.38	0.53	0.0	**21.3**	0.0	0.0	0.0	0.0
1.00	1.00	1.00	1.00	0.00	0.00	0.00	0.00	0.0	0.0	**62.2**	0.0	0.0	0.0
0.57	1.00	0.43	0.00	0.43	0.00	0.57	1.00	0.0	0.0	0.0	**34.4**	0.0	0.0
0.0	0.4	1.0	0.6	1.0	0.6	0.0	0.4	0.0	0.0	0.0	0.0	**34.4**	0.0
0.0	1.0	0.0	1.0	0.0	1.0	0.0	1.0	0.0	0.0	0.0	0.0	0.0	**27.7**
1.00	0.18	0.00	0.35	0.53	0.38	0.00	0.62	**-21.3**	0.0	0.0	0.0	0.0	0.0
0.18	1.00	0.35	0.00	0.38	0.53	0.62	0.00	0.0	**-21.3**	0.0	0.0	0.0	0.0
0.00	0.00	0.00	0.00	1.00	1.00	1.00	1.00	0.0	0.0	**-62.2**	0.0	0.0	0.0
0.43	0.00	0.57	1.00	0.57	1.00	0.43	0.00	0.0	0.0	0.0	**-34.4**	0.0	0.0
1.0	0.6	0.0	0.4	0.0	0.4	1.0	0.6	0.0	0.0	0.0	0.0	**-34.4**	0.0
1.0	0.0	1.0	0.0	1.0	0.0	1.0	0.0	0.0	0.0	0.0	0.0	0.0	**-27.7**

Maximum pure force and pure torque. Tilted 45 deg. Azimuth 22.5 degrees													
Thrust level (0 to 1) for each 22N Thruster								Fx	Fy	Fz	Tx	Ty	Tz
1	2	3	4	5	6	7	8	[N]	[N]	[N]	[N*m]	[N*m]	[N*m]
0.00	0.43	1.00	0.15	0.00	0.69	0.58	0.31	**18.9**	0.0	0.0	0.0	0.0	0.0
0.43	0.00	0.15	1.00	0.69	0.00	0.31	0.58	0.0	**18.9**	0.0	0.0	0.0	0.0
1.00	1.00	1.00	1.00	0.00	0.00	0.00	0.00	0.0	0.0	**62.2**	0.0	0.0	0.0
0.60	1.00	0.40	0.00	0.40	0.00	0.60	1.00	0.0	0.0	0.0	**34.4**	0.0	0.0
0.0	0.4	1.0	0.6	1.0	0.6	0.0	0.4	0.0	0.0	0.0	0.0	**34.4**	0.0
0.0	1.0	0.0	1.0	0.0	1.0	0.0	1.0	0.0	0.0	0.0	0.0	0.0	**41.0**
1.00	0.15	0.00	0.43	0.58	0.31	0.00	0.69	**-18.9**	0.0	0.0	0.0	0.0	0.0
0.15	1.00	0.43	0.00	0.31	0.58	0.69	0.00	0.0	**-18.9**	0.0	0.0	0.0	0.0
0.00	0.00	0.00	0.00	1.00	1.00	1.00	1.00	0.0	0.0	**-62.2**	0.0	0.0	0.0
0.40	0.00	0.60	1.00	0.60	1.00	0.40	0.00	0.0	0.0	0.0	**-34.4**	0.0	0.0
1.0	0.6	0.0	0.4	0.0	0.4	1.0	0.6	0.0	0.0	0.0	0.0	**-34.4**	0.0
1.0	0.0	1.0	0.0	1.0	0.0	1.0	0.0	0.0	0.0	0.0	0.0	0.0	**-41.0**

Maximum pure force and pure torque. Tilted 45 deg. Azimuth 45 degrees													
Thrust level (0 to 1) for each 22N Thruster								Fx	Fy	Fz	Tx	Ty	Tz
1	2	3	4	5	6	7	8	[N]	[N]	[N]	[N*m]	[N*m]	[N*m]
0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	**0.0**	0.0	0.0	0.0	0.0	0.0
0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.0	**0.0**	0.0	0.0	0.0	0.0
1.00	1.00	1.00	1.00	0.00	0.00	0.00	0.00	0.0	0.0	**62.2**	0.0	0.0	0.0
0.67	1.00	0.33	0.00	0.33	0.00	0.67	1.00	0.0	0.0	0.0	**35.5**	0.0	0.0
0.0	0.3	1.0	0.7	1.0	0.7	0.0	0.3	0.0	0.0	0.0	0.0	**35.5**	0.0
0.0	1.0	0.0	1.0	0.0	1.0	0.0	1.0	0.0	0.0	0.0	0.0	0.0	**75.7**
0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	**0.0**	0.0	0.0	0.0	0.0	0.0
0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.0	**0.0**	0.0	0.0	0.0	0.0
0.00	0.00	0.00	0.00	1.00	1.00	1.00	1.00	0.0	0.0	**-62.2**	0.0	0.0	0.0
0.33	0.00	0.67	1.00	0.67	1.00	0.33	0.00	0.0	0.0	0.0	**-35.5**	0.0	0.0
1.0	0.7	0.0	0.3	0.0	0.3	1.0	0.7	0.0	0.0	0.0	0.0	**-35.5**	0.0
1.0	0.0	1.0	0.0	1.0	0.0	1.0	0.0	0.0	0.0	0.0	0.0	0.0	**-75.7**

Table 15-7: Maximum pure torque and force for different RCS orientations

With the 15 deg azimuth configuration, the maximum forces are 21 and 62 N in the horizontal and vertical plane. This is believed to provide sufficient agility during the descent and final phase, pending further analysis in future phases of the activity.

For a hypothetical hovering phase, delta-v manoeuvres up to 1.5 m/s may need to be executed every 120 s (see section 15.2.5). Assuming a mass of 1257 kg, the necessary propellant to perform a 1.5 m/s manoeuvre is 0.66 kg for an Isp of 291 s. For the given thrust levels, this manoeuvre would require 88 or 30 seconds depending on which thrusters are used (horizontal or vertical). Based on this preliminary computations, it may be necessary to enlarge the constant time intervals of the hovering guidance scheme and its impact on the position accuracy.

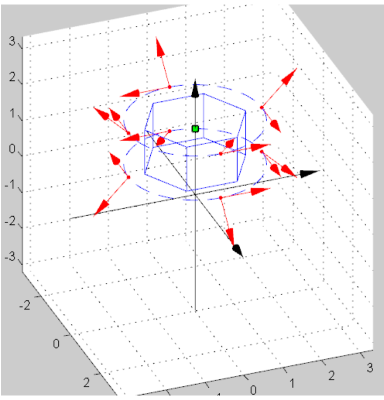
The last thing to assess is whether this 8 thruster configuration would be capable of performing the fine control needed during the landing phase. The minimum impulse bit (MIB) of the hydrazine 20N thruster is assumed to be 0.070 Ns, with an accuracy of 25%. Given this input, a similar table has been computed to obtain the minimum pure forces and torques for the 15 deg azimuth case (see Table 15-8). Note how the thrust level requested to each thruster (left part of the table) is never below 3.18E-3 s (22*3.18E-3 = 0.070 Ns).

Minimum pure force and pure torque. Tilted 45 deg. Azimuth 15 degrees														Min controllability (+25%)	
Thrust level (from 0 to 1) required for each of the 22 N Thrusters								Fx	Fy	Fz	Tx	Ty	Tz	Speed	Angular rate
1	2	3	4	5	6	7	8	[N]	[N]	[N]	[N*m]	[N*m]	[N*m]	[m/s]	[deg/s]
#####	5.82E-03	1.66E-02	3.18E-03	0.00E+00	1.03E-02	8.79E-03	6.33E-03	**0.355**	0.00	0.00	0.00	0.00	0.00	3.53E-04	-
#####	0.00E+00	3.18E-03	1.66E-02	1.03E-02	0.00E+00	6.33E-03	8.79E-03	0.00	**0.355**	0.00	0.00	0.00	0.00	3.53E-04	-
#####	3.18E-03	3.18E-03	3.18E-03	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00	0.00	**0.198**	0.00	0.00	0.00	1.97E-04	-
#####	7.18E-03	3.18E-03	0.00E+00	3.18E-03	0.00E+00	4.09E-03	7.18E-03	0.00	0.00	0.00	**0.248**	0.00	0.00	-	0.023
#####	3.18E-03	7.18E-03	4.09E-03	7.18E-03	4.09E-03	0.00E+00	3.18E-03	0.00	0.00	0.00	0.00	**0.248**	0.00	-	0.023
#####	3.18E-03	0.00E+00	3.18E-03	0.00E+00	3.18E-03	0.00E+00	3.18E-03	0.00	0.00	0.00	0.00	0.00	**0.088**	-	0.008
#####	3.18E-03	0.00E+00	5.82E-03	8.79E-03	6.33E-03	0.00E+00	1.03E-02	**-0.355**	0.00	0.00	0.00	0.00	0.00	-2.99E-04	-
#####	1.66E-02	5.82E-03	0.00E+00	6.33E-03	8.79E-03	1.03E-02	0.00E+00	0.00	**-0.355**	0.00	0.00	0.00	0.00	-2.99E-04	-
#####	0.00E+00	0.00E+00	0.00E+00	3.18E-03	3.18E-03	3.18E-03	3.18E-03	0.00	0.00	**-0.198**	0.00	0.00	0.00	-1.66E-04	-
#####	0.00E+00	4.09E-03	7.18E-03	4.09E-03	7.18E-03	3.18E-03	0.00E+00	0.00	0.00	0.00	**-0.248**	0.00	0.00	-	-0.015
#####	4.09E-03	0.00E+00	3.18E-03	0.00E+00	3.18E-03	7.18E-03	4.09E-03	0.00	0.00	0.00	0.00	**-0.248**	0.00	-	-0.015
#####	0.00E+00	3.18E-03	0.00E+00	3.18E-03	0.00E+00	3.18E-03	0.00E+00	0.00	0.00	0.00	0.00	0.00	**-0.088**	-	-0.004

Table 15-8: Minimum pure torque and force for 15 deg azimuth. Minimum controllability velocities and angular rates

Using the MCI input data presented in Figure 15-5 the minimum velocity and angular rate accuracy is presented in the last two columns of Table 15-8. They have been computing assuming a 25% uncertainty in the thrust level generated by the thrusters when operated close to their MIB. The result is that for the 15 deg azimuth RCS configuration the velocity accuracy which can be obtained is 0.03 cm/s, well beyond the minimum delta-v manoeuvres expected during hovering (0.2 m/s), the smallest maintenance QSO correction manoeuvres (in the order of 1 cm/s) and the lateral landing velocity requirement of 15 cm/s. In a similar way, the angular rate accuracy which can be obtained when operating the RCS is the order of 0.02 deg/s, which validates the conservative assumption of an initial angular rate of 0.1 deg/s at the beginning of the free-fall phase.

Finally, an alternative RCS configuration with 8 (x2 for redundancy) + 8 extra roll thrusters has been investigated (see Figure 15-7 and Table 15-9)



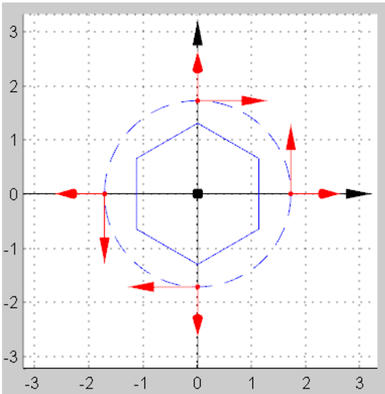


Figure 15-7: Configurations with 24 RCS: 8x2 (tilted 45 deg, azimuth 0 deg) + 8 roll thrusters.

Maximum pure force and pure torque. Tilted 45 deg. 8 RCS thrusters with 0 azimuth + 8 roll RCS																					
Thrust level (0 to 1) for each 22N Thruster																Fx	Fy	Fz	Tx	Ty	Tz
1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	[N]	[N]	[N]	[N*m]	[N*m]	[N*m]
0.00	0.00	1.00	0.00	0.62	0.00	0.38	0.00	1.00	0.00	1.00	1.00	0.00	0.00	0.00	1.00	**55.9**	0.0	0.0	0.0	0.0	0.0
0.00	0.00	0.00	1.00	0.00	0.62	0.00	0.38	0.00	1.00	1.00	1.00	0.00	0.00	1.00	0.00	0.0	**55.9**	0.0	0.0	0.0	0.0
1.00	1.00	1.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.0	0.0	**62.2**	0.0	0.0	0.0
0.78	1.00	0.00	0.00	0.00	0.00	0.78	1.00	0.00	1.00	1.00	0.00	1.00	0.00	0.00	1.00	0.0	0.0	0.0	**61.4**	0.0	0.0
0.0	0.0	1.0	0.0	1.0	0.0	0.0	0.0	1.0	1.0	1.0	0.0	0.0	0.0	0.0	1.0	0.0	0.0	0.0	0.0	**61.4**	0.0
0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0	1.0	0.0	1.0	0.0	1.0	0.0	0.0	0.0	0.0	0.0	**151.4**
1.00	0.23	0.00	0.00	0.00	1.00	0.23	0.00	0.00	1.00	0.93	0.00	0.56	1.00	0.51	0.00	**-55.9**	0.0	0.0	0.0	0.0	0.0
0.00	1.00	0.00	0.00	0.00	0.38	0.00	0.62	1.00	1.00	0.00	1.00	1.00	0.00	0.00	0.00	0.0	**-55.9**	0.0	0.0	0.0	0.0
0.00	0.00	0.00	0.00	1.00	1.00	1.00	1.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.0	0.0	**-62.2**	0.0	0.0	0.0
0.78	0.00	0.00	1.00	0.00	1.00	0.78	0.00	1.00	1.00	0.00	0.00	0.00	0.00	1.00	1.00	0.0	0.0	0.0	**-61.4**	0.0	0.0
1.0	0.0	0.0	0.8	0.0	0.8	1.0	0.0	0.0	0.0	1.0	1.0	1.0	1.0	0.0	0.0	0.0	0.0	0.0	0.0	**-61.4**	0.0
0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	1.0	0.0	1.0	0.0	1.0	0.0	1.0	0.0	0.0	0.0	0.0	0.0	0.0	**-151.4**

Table 15-9: Maximum pure torque and force for 24 RCS configuration (option)

This configuration increases the control authority in all 6 degrees of freedom, especially in the roll torque (151 Nm) which becomes closer to the standard Fregat roll torque authority (estimated in 168 Nm) and ensuring therefore the controllability of the stack (propulsion module+lander) during the main engine burns.

15.3 Baseline Design

Two possible options can be envisaged for the GNC descent approach: direct descent and body-fixed hovering. More detailed GNC analysis are needed to properly trade-off the best solution, since both options have their own advantages and drawbacks as explained in section 15.2.5. For the purpose of the delta-v budget elaboration the direct descent option has been retained, pending confirmation that the landing accuracy and the operational constraints imposed by this solution, are acceptable.

A preliminary delta-v budget is presented in Table 15-10, resulting in a total amount of 350 m/s including margins for the Phobos close proximity operations and descent and landing. The numbers are mostly based on the results presented in RD[22] including conservative margins for control errors and efficiency factors to take into account that the RCS engines are tilted 45 degrees. For the purpose of this CDF study this delta-v budget is believed to be sufficiently accurate, but a more refined results based on duration of each of the phases, number of descent rehearsals and aborts, delta-v consumption based on GNC montecarlo simulations, etc. will be elaborated as part of future activities (RD[28]).

Mission Phases	Manoeuvre	Engine	DV [m/s]	Margin [%]	Total DV [m/s]
Phobos Close Proximity	Safe Mode	Low Thrust	1.00	100.00	2.00
	Attitude Control	Low Thrust	2.00	100.00	4.00
	Trailing -> Heading Orbit	Low Thrust	6.00	5.00	6.30
	QSO	Low Thrust	20.00	5.00	21.00
	High Latitude Observation	Low Thrust	35.00	5.00	36.75
	Movement to Safe Orbit	Low Thrust	20.00	5.00	21.00
	Station Keeping	Low Thrust	12.00	5.00	12.60
	RW Off-loading	Low Thrust	2.00	100.00	4.00
	Fly-bys (3)	Low Thrust	40.00	5.00	42.00
	Provision for tilted thrusters	Low Thrust	33.50	5.00	35.18
	Total Delta-V PCP				**184.83**
Descent and Landing	Direct landing, thrusters 45° tilted	Low Thrust	150.00	10.00	165.00
	Total Delta-V D&L				**165.00**
Surface	**Sampling / Hold-down Force (*)**	Low Thrust			0.00
	Total Delta-V Surface (*)				**0.00**

Table 15-10: Delta-v budget for close proximity, descent and landing at Phobos

In terms of hardware, a set of redundant sensors are envisaged for the mission: 2 star trackers, 2 IMUs and 2 sun sensors to provide classical gyro-stellar attitude determination; two altimeters and two wide angle camera for final descent and landing and the use of the science narrow angle camera to aid in the navigation at large distances from Phobos.

In terms of actuators, 4 reaction wheels will be used for attitude control during close proximity operations. In addition they will most likely be needed during the free-fall phase in order to reduce the angular displacement with respect to the gravitational acceleration at touchdown (see section 15.2.3). For attitude and orbit control, it is envisaged to use 16 x 20N hydrazine thrusters (8 redundant thrusters) canted 45 degrees with respect to the longitudinal axis and 15deg in azimuth. This type of configuration has shown to provide pure torque and force in all directions with the necessary authority and precision. An alternative option of 16 + 8 roll thrusters may be needed to increase roll authority. This option would only be needed to compensate a big roll perturbation torque generated when firing the main Fregat-derived engine (which is not yet confirmed at the time of writing this report).

15.4 List of Equipment

Element 1	-			MASS [kg]				Power (W)
Unit	Unit Name Click on button above to insert new unit	Part of custom subsystem	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin	Ppeak
1	AA STR		2	1.500	To be developed	20	3.6	7.7
2	European IMU (Astrix1090 + QA3000)		2	3.000	To be developed	20	7.2	12.0
3	RW RSI 12/75-60		4	4.800	Fully developed	5	20.2	20.0
4	Radar Altimeter		2	2.000	To be developed	20	4.8	8.0
5	Sun Sensor TNO		2	0.050	Fully developed	5	0.1	0.0
6	Wide Angle Camera (2OH+1EU)		2	2.000	To be developed	20	4.8	4.0
-	Click on button below to insert new unit							
	SUBSYSTEM TOTAL		6	36.3		12.0	40.7	51.7

Table 15-11: GNC equipment summary list.

15.5 Technology Requirements

The following technologies are required and would need to be further developed for the Phobos Sample Return missions:

Equipment and Text Reference	Technology	Suppliers and TRL Level	Additional Information
Wide Angle Camera and vision-based navigation filter	APS sensor, image processing & filtering for vision-based navigation	Airbus D&S, TAS-I. TRL 5	ESA development. NPAL & VisNAV activity.
Range altimeter	Radar or laser	EU industry. TRL 3/4	ESA development, RD[27]
IMU	European IMU	Airbus D&S, TRL 3	Breadboard ready in 2014 (TRL 4-5)
GNC for proximity operations, descent and landing.	Absolute and relative navigation, guidance and control.	TRL-3/4	RD[28]

16 POWER

16.1 Requirements and Design Drivers

The power system shall provide electrical power to the subsystems and payload throughout all the mission phases.

The design of the electrical power subsystem for spacecraft is subject to several mission specific factors. The main sizing parameters and design drivers include average and peak power demands, mission duration and orbit profile. This drives the selection of the energy sources and of the power management and distribution.

For Phobos Sample Return, the analysis and sizing of the power system is made more complex than for more conventional studies due to the fact that the Landing Module has to stay on the surface of Phobos for an extended period.

The mathematical model built for the MMSR-A5 study was re-used to calculate the amount of energy gathered by the solar panels of the spacecraft during and after landing.

The other phases of the mission are more classical, with a GCP phase (orbit around Phobos) lasting several months, and where eclipse durations vary from 0 to 60 minutes.

16.2 Architecture Trade-Offs

16.2.1 Regulated vs. Unregulated

Some trade-offs exist for deciding on the power systems architecture to be used. The most obvious one is the regulated vs. unregulated bus. Compared to a classical regulated bus, there is a mass benefit on the power system when having an unregulated bus since the BCR and BDR are not implemented. There is also a small impact on the battery size since inefficiencies linked to BCR and BDR are not present anymore (direct battery power transfer). The solar array area is also marginally reduced, because of the higher efficiency of the solar array to battery path. Finally, lower power dissipation occurs on-board during battery charge and discharge, which may facilitate thermal management for the power system.

Of course disadvantages also exist for the unregulated bus, the main ones being that the protection levels for the distribution are more difficult to set due to the varying bus voltage, there are some more EMC issues for the same reason, the heater power levels vary with the bus voltage if no constant current drivers are implemented, and the secondary power converters in the loads will be sized for the whole bus voltage range, resulting in somewhat less efficient converters than for the regulated bus option.

For Phobos Sample Return, the battery will potentially be discharged down to 20 % SoC as part of the nominal mission (end of Phobos surface operations) and it means large excursions of the bus voltage for an unregulated bus. To be on the safe side for this mission, a regulated bus was chosen. This trade-off should only have a low impact on the systems mass anyway and can be reopened later in the study.

16.2.2 MPPT vs. S3R

The other trade-off that is often performed is about the SAR converter: MPPT or S3R. The first is heavier, less efficient but is able to extract all the solar array power available under a large range of conditions (BoL, EoL, different SAA, etc.). The latter is simpler, lighter, cheaper and more efficient but is very rigid in the way power extraction is performed. In other words, an S3R based SAR will be very efficient for a narrow range of conditions but will lose its advantages if flexibility is required. For Phobos Sample Return, flexibility is required in the sense that the sun to spacecraft distance varies from 1 to 1.65 AU, resulting in a 2.7 to 1 solar flux ratio. Because of this the operating temperature range of the solar array is large, and as a result its MPP varies significantly. Furthermore, the SAA will also vary significantly during and after landing, inducing even more temperature variations. MPPT is a must have on this mission and is assumed for the analysis.

16.3 Power Budgets

The power budget for Phobos Sample Return is shown in the following table.

MISSION PHASE	LAUNCH	GCP - sun	GCP - eclipse	LCP - sun	LCP - eclipse	Descent	TxEarth	Night (Standby)	Day
LM	497	984	1001	675	758	696	471	256	322
Losses (LCL, harness)	15.1	35.0	41.5	28.8	38.5	29.8	18.1	13.6	11.1
OBDH S/S	36	55	55	55	55	55	35	31	31
EPS S/S	24	21	29	21	29	21	21	21	21
COMMS S/S	39	158	158	158	158	145	171	0	0
Thermal S/S	0	115	115	115	115	115	150	150	150
AOCS	0	103	103	108	161	161	0	0	0
MECHANISM S/S	0	0	0	0	0	0	0	0	39
PROPULSION S/S	3	3	3	45	45	58	0	0	0
PAYLOAD	0	36	36	36	36	0	0	0	18
PM	300	300	300	0	0	0	0	0	0

Table 16-1: Phobos-SR power budget: LM

The harness (PCDU to loads) losses are accounted for in this budget, taken as 2 % of the load power. Please note that the power budget presented is at the SAR output, and accounts for distribution losses occurring in the PDU switches (1%). The other losses occurring in the SAR or solar array diodes and harness are taken into account in the solar array sizing process, and do not appear in the power budget. Furthermore, it is assumed that the ERV is powered by its own solar array and battery. Hence, no power needs to be provided by the LM. If the LM needs to provide additional power for the ERV, this might lead to an increase of the solar array size and would need to be assessed in potential follow-up studies. Note that it is assumed that the ERV is self-standing in power during the complete mission due to the orientation of its panels and the size of its batteries. This is confirmed in the ERV description section.

16.4 Power System Performance on Phobos

In order to conduct the sizing of the power system it was necessary to model the Phobos environment in terms of solar illumination. A MathCad model was built for the MMSR-A5 (Phootprint) study and was modified for this study.

16.4.1 Seasonal Variations

First there is the variation in Sun to Mars distance versus time, due to Mars orbit eccentricity around the Sun. Figure 16-1 shows this variation, together with two possible landing dates. Note that the time of landing is driven by this parameter, in order to minimise impact on solar array size. The candidate landing dates mean that a Sun to Mars distance of 1.4 AU can be considered for the sizing, while allowing time slots of 4 month for landing and operations.

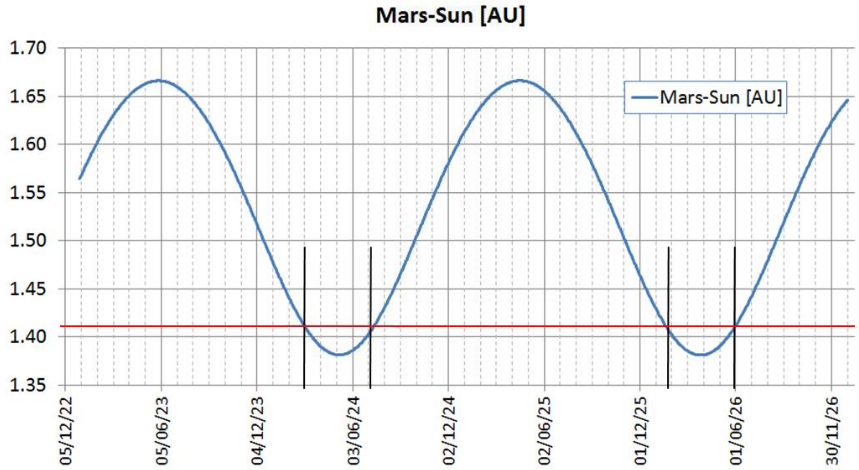


Figure 16-1: Sun to Mars (Phobos) distance vs. time

Phobos orbits nearly in Mars equatorial plane with a period of 7 hours and 39 minutes. This orbit is slightly eccentric but it will be assumed completely circular to simplify the model. Phobos rotation period is synchronous with its orbital period. Although Phobos is highly non spherical, for the sake of this study it has been assumed to be perfectly spherical. The impact of this simplification is discussed later. The Martian equatorial plane is tilted 25° with respect to the ecliptic, and this implies seasonal variations over a Martian year (24 months). The following plot shows the Sun latitude over Mars as a function of the date.

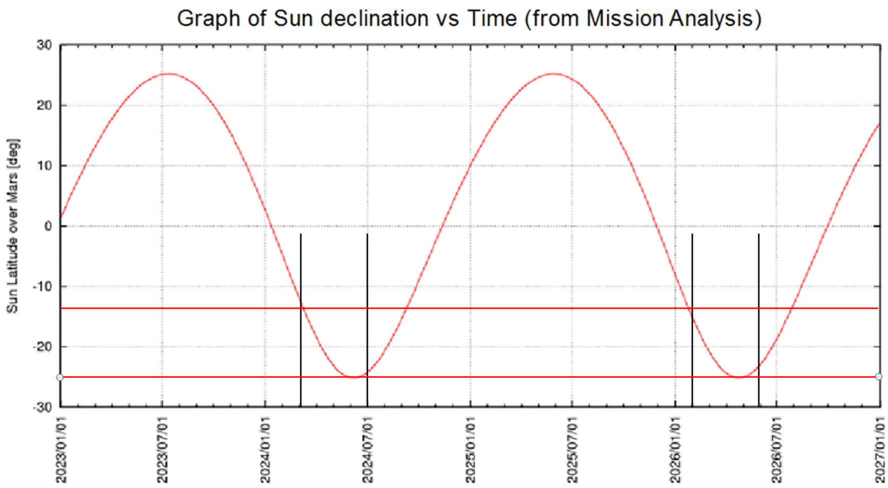


Figure 16-2: Sun Latitude over Mars/Phobos versus Date

These seasonal variations mean that not all of the surface might be accessible because of possible permanent night at high or low latitudes. The following figures show examples of how the Moon is illuminated over a Martian year. The lander reference frame is shown, with a landing site on the equator.

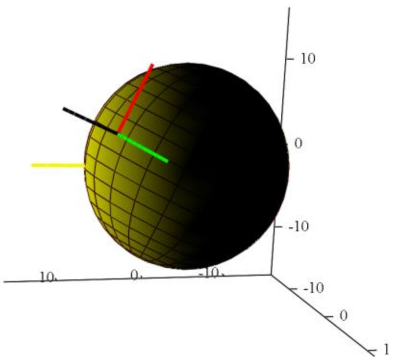


Figure 16-3: Southern Summer, June 2024

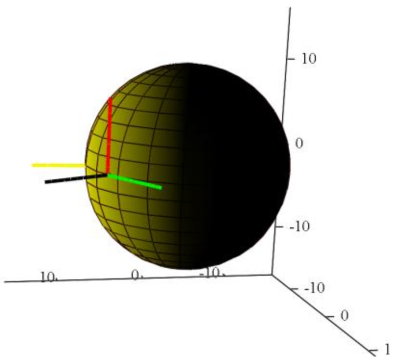


Figure 16-4: Equinox, October 2024

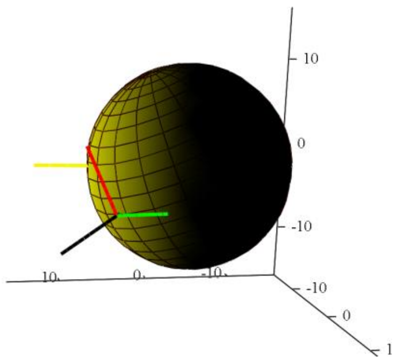


Figure 16-5: Northern Summer, June 2025

The sun latitude variations induce variations in the day durations as Mars orbits the Sun. The next figures show this trend. The landing latitude also plays a role, with possible permanent nights or days at the poles, as on Earth.

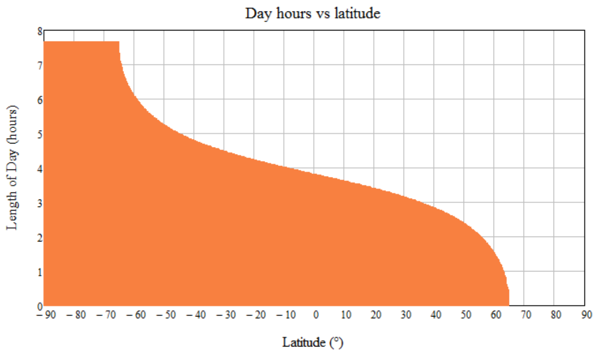


Figure 16-6: Day Duration vs Latitude, June 2024

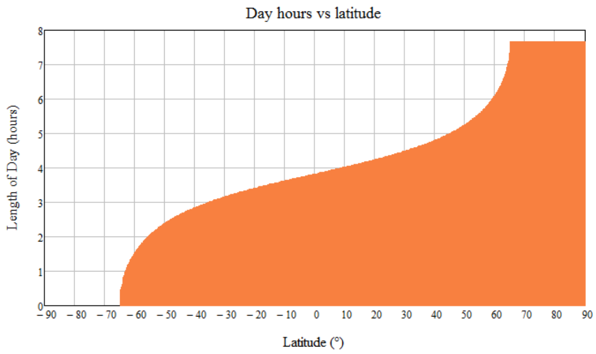


Figure 16-7: Day Duration vs Latitude, June 2025

Please note that during the Equinox (October 2024), the day duration is half of the Phobos rotation period and is invariant with the latitude.

16.4.2 Sun Azimuth, Elevation and SAA

Figure 16-8 shows the angles that are referenced in this chapter. The latitude and longitude define the landing site, whereas the elevation and azimuth define the sun position in the local sky.

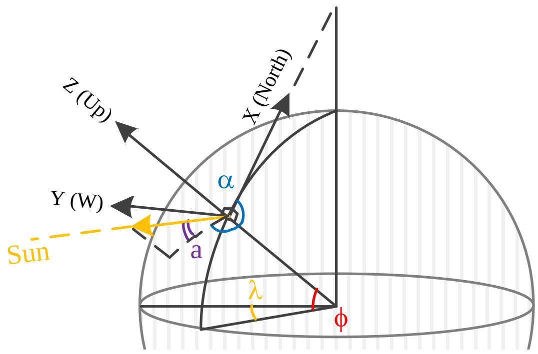


Figure 16-8: Definitions of Longitude (λ), Latitude (ϕ), Elevation (a) and Azimuth (α) used in this chapter

To calculate the amount of solar power reaching the SA at any time, it is necessary to derive the local Azimuth and Elevation angles for landing sites situated at different latitudes. The following figures show the Sun position in the local sky (Figure 16-9 and Figure 16-10) as well as the Elevation versus Time (Figure 16-11 and Figure 16-12). The longitude of the landing site is not considered in the study as it does not play a significant role, apart from shifting all the curves left or right (in time).

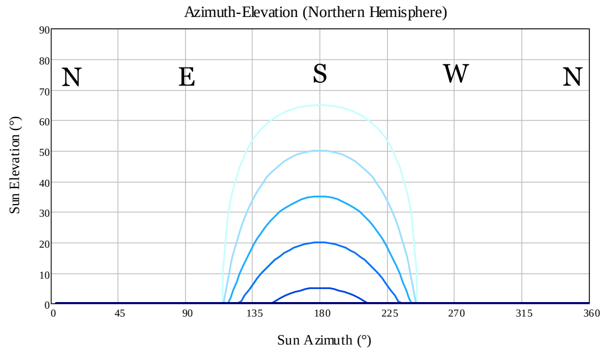


Figure 16-9: Sun Elevation versus Azimuth, for different landing site latitudes (0° to +90° in steps of 15° - May 2024)

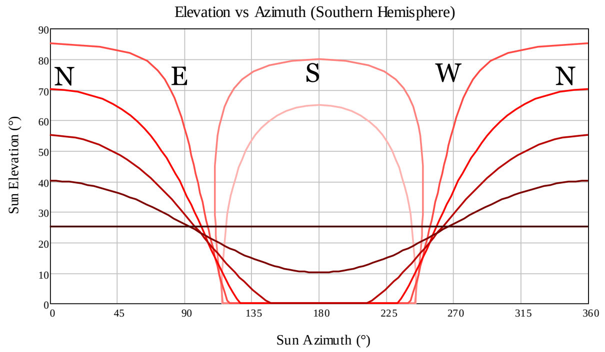


Figure 16-10: Sun Elevation versus Azimuth, for different landing site latitudes (0° to -90° in steps of 15° - May 2024)

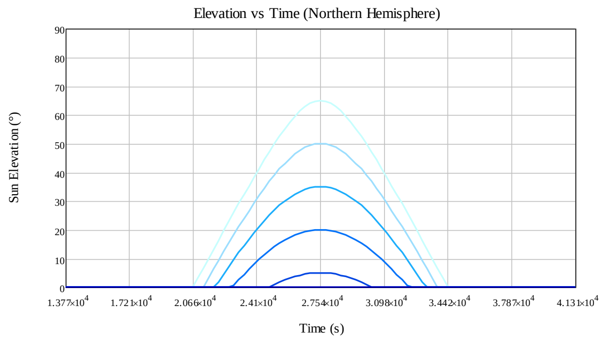


Figure 16-11: Sun Elevation versus time, for different latitude landing sites (0° to +90° in steps of 15° - May 2024)

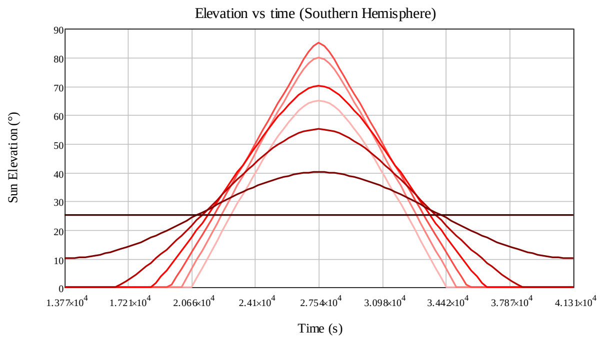


Figure 16-12: Sun Elevation versus time, for different latitude landing sites (0° to -90° in steps of 15° - May 2024)

Once the elevation is known, the SAA may be derived. The lander will be equipped with two SA wings, without SADM. This is a system level decision, owing to the fact that SADM capable of withstanding the potential landing loads are deemed too costly and too long to develop.

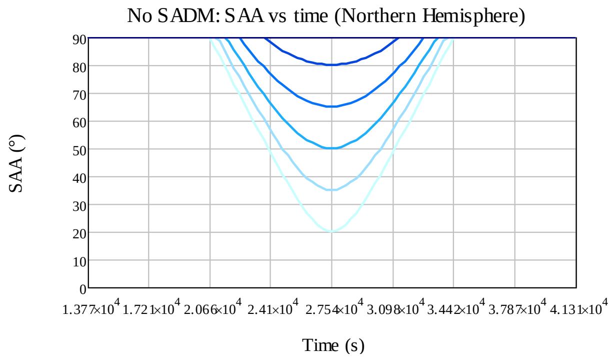


Figure 16-13: SAA vs. time, landing site latitudes of 0° to +90° in steps of 15° - May 2024

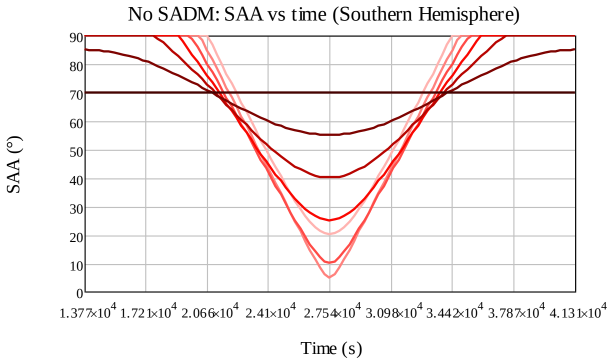


Figure 16-14: SAA vs. time, landing site latitudes of 0° to +90° in steps of 15° - May 2024

16.4.3 Energy Production

Degradation Factors for Cell Current are as follows:

Cell Parameter	Degradation Factor	Direct/Rdm
Cell Mismatch	0.99	*random*
Calibration	0.97	*random*
Cover Glass	0.985	*direct*
Pointing error	1	*direct (°)*
UV degradation	0.99	*direct*
Micrometeorites	0.99	*direct*

Table 16-2 : Additional Solar Cell degradation factors

The following additional assumptions have been made for the sizing:

- 10°C temperature margins for SA thermal model
- Cell active area to panel area ratio of 0.85
- Panel mass of 4.5 kg/m².
- Single string failure tolerant design
- 30 % AsGa Triple Junction solar Cells from Azur Space
- SA harness drop 1 V, diode drop 0.8 V
- EoL Radiation dose given as ~ 5E14 1MeV electron fluence. Not from analysis, may be reviewed in future stages of the study.

Once the SAA is known it can be used with the power vs. SAA “transfer function” of the solar array. This was calculated using a model that takes into account ageing, radiation effects, temperature effects, SAR efficiency (95%) and is shown in the following figure.

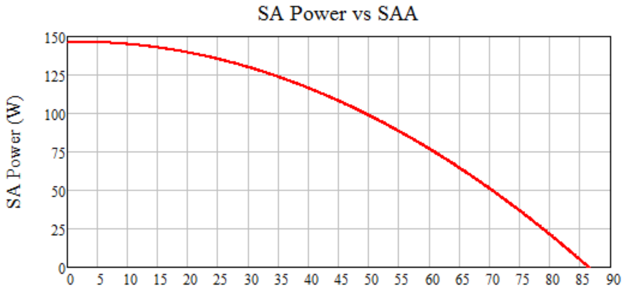


Figure 16-15: SAR output power vs SAA, for 1 m², June 2024 (1.4 AU)

From the SAA curves and the curve shown on Figure 16-15, the daily energy production per m² could be derived as a function of the landing latitude and seasons.

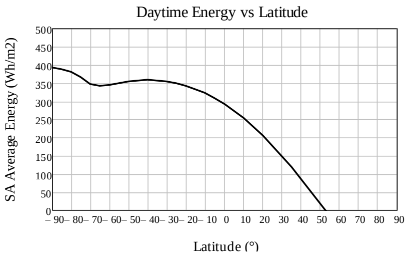


Figure 16-16: Daylight SAR energy per m², June 2024 (1.4 AU)

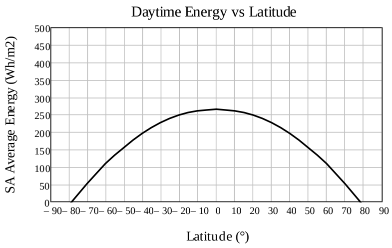


Figure 16-17: Daylight SAR energy per m2, October 2024 (1.52 AU)

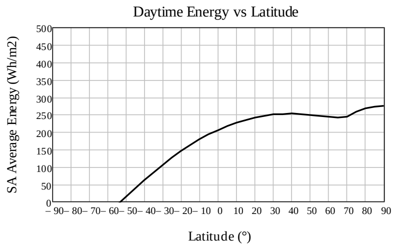


Figure 16-18: Daylight SAR energy per m2, June 2025 (1.67 AU)

16.5 Sizing

The sizing of the battery, solar array and PCDU for the lander was performed for GCP, descent, and surface operations. The Deimos orbiting phase was not studied directly as it is not a sizing case for the power systems. To check the sizing, a simple energy model was used, as shown in the following figure.

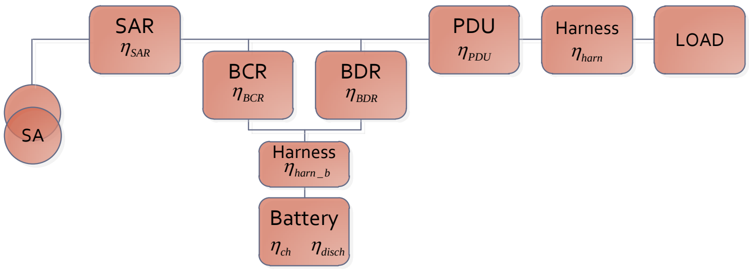


Figure 16-19: Simple model used for energy balance analysis

For GCP, a 459 minute orbit period was considered, together with an eclipse duration of 60 minutes. For surface operations, the mathematical model was used to derive the lander daily energy needs (for 10.5 m² SA), and to compare them with the daily energy production vs. landing latitude.

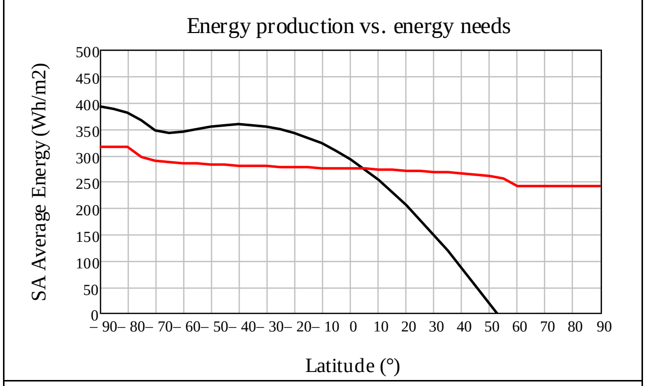


Figure 16-20: Daily energy production (black) vs. daily energy needs (red). Wh/m2/day

Note that the above plot corresponds to the 2024 winter solstice, while the baseline scenario foresee to land next to the 2026 winter solstice, so the actual energy production plot would be very similar. It is then shown that with 10.5m2 solar panel, landing on the Southern hemisphere ensures a balanced power budget. If the mission only targets 20% surface accessibility, then the solar panel size could even be reduced.

However, one aspect to be carefully addressed in future phases is the possible shadowing of part of the solar panels by the spacecraft: this could lead to reduce the accessibility from power point of view.

The descent is assumed to last for 3 hours. For the first 2 hours, the panels are illuminated with a SAA of 60 deg. For the last hour, only the battery is used. Here again this is considered to be very conservative.

Parameter	GCP	Surf Ops
Battery Cycled Energy	1119 Wh	1093 Wh
Required SAR Power	1207 W	750 W avg
Required SA Area	9.81 m^2(1)	10.5 m^2

Table 16-3: GCP and Surface operation power/energy requirements

(1) Nominal operations up to 1.52 AU possible

Looking at Figure 16-16 and Figure 16-20, for surface operations the daytime energy production required in order to achieve energy balance translates into a possible landing latitude range of [-90°, 5°].

Hereafter is the list of parameters used for the sizing.

Eff. Factor	Value	Comments
η_{SAR}	0.95	Converter efficiency
η_{disch}, η_{ch}	0.975	Battery power losses during charge/discharge
η_{harn}	1	Included in power budget
η_{harn_b}	0.995	Battery harness losses
η_{PDU}	1	Included in power budget
η_{BDR}, η_{BCR}	0.95	Regulated bus

Table 16-4 : Efficiency factors used in power systems model

Since 10.5 m² are needed for surface operations, a check was made for the descent mode. 10.5 m² with a SAA of 60° will produce ~ 785 W. Since the power requirement for this mode is about 700 W, the battery will not discharge for the first two hours, and will only be needed for the last hour. So the energy requirement during descent, for the battery, is 700 Wh.

Phase	GCP	Descent	Surface	
Energy Req	1119	700	1093	Wh
DoD	50	50	50	%
Cap loss	15	15	15	%
BoL req	2633	1647	**2572**	Wh
Mass	25.1	15.7	**24.5**	kg

Table 16-5: Battery sizing for three mission phases

Table 16-5 shows that the battery required for the mission is around 25 kg and 2.6 kWh.

16.6 Further Work

Here is a list of simplifications that have been made in order to make the analysis possible in the timeframe of the study. They should be further investigated during future stages but do not represent any show stopper for the mission.

1. Phobos landing: perfect sphere assumption

Phobos is far from being a perfect sphere and therefore the local topography should be included in the power analysis in order to derive a map of possible landing sites.

2. Phobos landing: uncertainty on lander tilt

A realistic lander tilt should be accounted for, as it will have an impact on daily energy production. 10° was assumed in the study.

3. Phobos landing: Mars eclipses

No Mars eclipse was considered during the landing phase on Phobos. It was assumed that any Mars eclipse occurs during the Phobos night. This has impacts on the possible range of longitudes that can be accessed for landing and should be investigated further. It is estimated that at least 315° of longitude range can be accessed out of 360° during Equinox season (worst season for Mars eclipses). This adds to the latitude constraints already identified.

16.7 List of Equipment (LM)

The following mass margins have been applied:

- PCDU: 10 %
- Battery : 10 %
- Solar Panels : 20 %

Equipment Name	Quantity	Dimensions	Mass (with margin)
PCDU	1	216 x 265 x 230 mm	18.7 kg
Battery	1	345 x 220 x 145 mm	27 kg
Solar Panels	1	10.5 m^2	56.7 kg

Table 16-6: Lander Equipment List

The Solar Panels mass differs by approximately 5 kg from the same mass in the LM mass budget and the equipment list. This is due to a late design change update but its effects are well within the uncertainties typical at this stage.

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17 DATA HANDLING

17.1 Requirements and Design Drivers

17.1.1 Functional

The command and data handling shall provide the capability to:

- Transfer information within the spacecraft (e.g. via data bus, point to point lines)
- Exchange information with the external environment (i.e. with EGSE, the launcher, ground segment)
- Process information so as to meet the mission objectives (i.e. provide processing resources to execute computer programs)
- Retain information for allocated periods of time so as to meet the mission objectives (e.g. store telemetry data during communication interruptions or context data in auxiliary (mass) memory)
- Maintain and disseminate timing information on-board.

17.1.2 Dependability

- A redundant functional path shall be triggered and used upon any failure, independently from ground control
- Failure tolerant design shall be applied to command and data handling whenever a potential for catastrophic or critical consequences exists.

17.1.3 Programmatics

- For data handling no specific new development should be required
- Off the shelf equipment shall be used wherever possible
- A cost and risk minimisation mission design approach shall be followed.

17.1.4 Design Drivers

For this mission, the main criteria for the command and data handling design are:

- Mass, volume and electrical power constraints
- Environmental conditions (e.g. temperature, ionizing radiations, vibrations or shocks, electromagnetic compatibility)
- Performance objectives (time criticality, dependability)
- The volume of information to be stored on-board, which depends on the rate of data generated on-board (by the orbital payload), the telemetry (and telecommand) data rates and the schedule of communication with the ground segment
- Programmatic objectives: see above

17.2 Assumptions and Trade-Offs

Based on the Phootprint study, it is assumed that the maximum amount of scientific and HK data stored onboard the spacecraft over the mission is lower than 128 Gbit.

It is assumed that a single failure tolerant design is sufficient for the mission.

It is assumed that the interfaces between European and Russian modules of the mission shall be minimised. Ideally, the different modules should operate independently with minimum cooperation.

17.3 Baseline Design

Neither the amount of data to be stored onboard, nor the required performance for the processor is expected to be really demanding. Therefore, the proposed design for the data handling avionics of the Landing Vehicle is based on a centralized architecture, with a single unit implementing the OBC, MMU and IO functionality.

This centralized solution will minimise the mass and power consumption of the design, while providing enough capabilities to fulfil the mission requirements.

In order to reduce the complexity of the design, the type and number of IO interfaces shall be minimised. For a mission with this data handling requirements CAN bus at 1Mbps would be an ideal candidate for both platform and payload bus. Discrete interfaces such as PackeWire, SpaceWire, UARTs, parallel interfaces as well as HV-HPC, BSM, TSM and similar shall be minimised.

The unit will likely contain the following internal modules:

- Power Conversion Module
- Reconfiguration Module: FDIR, Safeguard Memory and OBT
- Processor Module. Based on fault tolerant processor (e. g. LEON-FT series)
- Mass Memory Module. NAND Flash based
- IO Module: Platform and Payload bus, TTC, discrete sensors and actuators.

The design is based on a full redundant architecture. During the non-critical phases of the mission, the unit shall operate in cold redundant configuration to reduce the power consumption. While, it shall operate in hot redundant configuration when fast reconfiguration capability is required (e.g. landing phase).

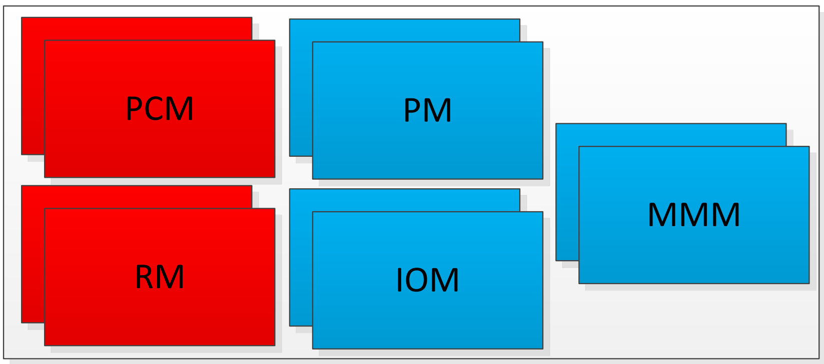

Figure 17-1: Cold Redundant configuration

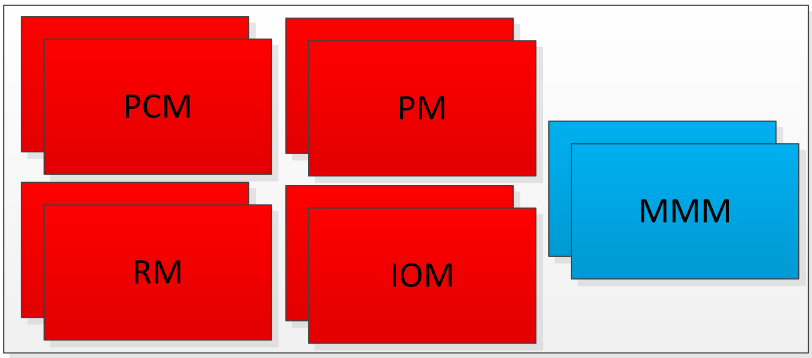

Figure 17-2: Hot Redundant configuration

17.4 List of Equipment

The following tables give the mass, dimensions and power requirements of the equipment.

Element 1	-			MASS [kg]			
Unit	Unit Name Click on button above to insert **new unit**	Part of custom subsystem	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
1	OBC + MM		1	11.0	To be modified	10	12.1
2			0		To be modified	10	0.0
-	Click on button below to insert new unit						
SUBSYSTEM TOTAL			1	11.0		10.0	12.1

Table 17-1: Mass budget

Element 1	-			DIMENSIONS [m]			
Unit	Unit Name Click on button above to insert **new unit**	Part of custom subsystem	Quantity	Shape	Dim1 Length	Dim2 Width or D	Dim3 Height
1	OBC + MM		1		0.4	0.3	0.1
2			0		#REF!	#REF!	#REF!
-	Click on button below to insert new unit			-			
SUBSYSTEM TOTAL			1				

Table 17-2: Dimensions

Element 1	-				-
Unit	Unit Name Click on button above to insert **new unit**	Part of custom subsystem	Quantity	Ppeak	Pon
1	OBC + MM		1	60.0	30.0
2			0		
-	Click on button below to insert new unit				
SUBSYSTEM TOTAL			1	60.0	30.0

Table 17-3: Power budget

17.5 Options

N/A

17.6 Technology Requirements

The technology referred in this document has a TRL > 5 at the time of writing this report.

The equipment units are expected to be readily available from industrial sources in Europe with possible delta qualification in case of modification.

The total cost estimate for command and data handling is expected to remain in the same order of magnitude irrespectively of the selected design.

Technology activities related to on-board computers integrating mass memory are programmed for the future.

18 TELECOMMUNICATIONS

18.1 Requirements and Design Drivers

The following points have been considered as design drivers:

- Single Point failures shall be avoided
- Redundancy shall be provided
- The re-use of fully developed space qualified hardware when possible.

18.1.1 TT&C Functionalities

- It shall receive and demodulate the uplink signal and transmit the data to the on-board data handling
- It shall modulate and transmit the telemetry
- It shall provide ranging and DOR capabilities (REQ LM-250)
- REQ LM-260 states that real-time information shall be sent to Earth during the descent to Phobos
- The maximum distance to Earth is 2.4 AU
- Link budget margins shall comply with the ECSS-E-ST-50-05C RF and Modulation Standard. For the nominal case the margin shall be higher than 3 dB.

18.2 Assumptions and Trade-Offs

18.2.1 Frequency Selection

The X-band Deep Space allocation is selected.

It shall be noted that S-band and Ka-band Deep Space allocations are also available but the X-band has been selected for compatibility with the ground station network (currently the ESA Deep Space Network implements X-band uplink and downlinks in all stations and only Ka-band reception in DSA2 and DSA3) and the availability of fully developed X-band hardware.

In addition REQ LM-220 indicates that the system shall be compatible with the Russian ground station, which also implements the X-band allocation.

18.2.2 Ground Station Assumptions

As per REQ LM-220 the lander shall be compatible with the ESA and ROSCOSMOS Deep Space stations.

It shall be noted that the ROSCOSMOS ground station performance has not been provided.

It is believed that the Rosocosmos ground station provides different polarisations for the uplink and downlinks while the proposed on-board architecture is based on antennas that transmit and receive in the same polarisation. It is however expected that in the frame of the Exomars 2018 mission this is discussed and a way forward agreed.

No EIRP performance has been provided for the G/T, A G/T performance of 59.5 dBK at 5 degrees has been assumed, but needs confirmation.

One ground station shall be preferred [REQ NI-160], the communications window assumed for the cruise, fly-by and up to landing phase is 8 hours while 2 hours of visibility per Phobos day are considered during the surface phase.

18.2.3 Data Generation and Transmission

Worst case data production for the analysis has been considered:

- Far Global Characterisation; data generated 12.8 Gb
- Deimos Quasi Satellite Orbit; data generated 26.6 Gb
- Phobos Quasi Satellite Orbit; data generated 63.8 Gb
- Fly-by; each fly-by will generate 9.17 Gb, there are 3 fly-bys generated data is 27.5 Gb
- Sampling Site Imaging data volume: 3 Gb (this data volume includes the Russian camera data generation)
- Sampling Site Characterisation; 3 Gb
- Close – up imaging before and after sampling; 1.2 Gb

For the Housekeeping data generation 1250 bps has been assumed, which is considered on the high side, no data compression has been considered.

A Phobos day will have 3.5 hours of day and night duration.

The instrument data is compressed at the source, each instrument will provide compression.

18.2.4 Communication Strategy Definition

Communications will be performed through the LGA during the LEOP phase up to a distance in which the HGA does not exceed the power flux density and in safe mode. The HGA will be used as soon as possible but it shall be noted that the pointing accuracy is critical to the performance of the link.

Three LGAs have been accommodated in the composite spacecraft to provide almost full coverage during the different phases. The location of the antennas is critical since there is the potential of interference from other elements, in particular the solar array panels, the plume from the thrusters, and the spacecraft structure.

Specific investigations are requested to define the envelope of Earth apparent angles in order to investigate if the inclusion of an MGA is beneficial.

A mechanical steerable HGA with two degrees of freedom is needed to transmit the data generated during the critical communications missions phases; fly-by, QSO and nominal operations on the surface of Phobos.

The communications equipment is accommodated on-board the lander which will remain on the surface of Phobos once the ERC has departed. This will allow the transmission of the stored data back to Earth.

During the descent and landing, though the g-forces seem to be compatible with the mechanism design, it is recommended that the HGA is stowed close to the spacecraft panel to avoid any stress. The communications during this phase will be performed through the LGA to ensure a reliable (attitude independent in case of non nominal mission behaviour) link with ground (see below).

18.2.4.1 Descent and Landing Communications

REQ LM-260 states that real-time information shall be sent to Earth during the descent to Phobos. The communications signal during this phase will suffer from strong signal dynamics however these are not expected to be as severe as for the Mars missions. The free-fall and the firing of the thrusters will be the major cause of the dynamics experienced by the composite.

The LGA as stated in the previous section has been selected to ensure an attitude independent link with ground since it will provide a quasi omni-directional pattern. However, the use of a low gain antenna implies that the EIRP will be low in particular in the directions far from the boresight.

Current investigations indicate that with a 35m G/S (ESA Deep Space Antenna) and the on-board technology selected, the required C/No to receive specific MFSK tones is not fulfilled at the ground station receiver under all conditions. However, the Russian ground station with a better performance could receive the signal and needs to be further investigated. This would also be the case with the NASA 70m dish.

The implementation of MFSK tones will be a more simple system, each tone represents an event however the reception of HK data should also be investigated since it will provide more detailed information on the events occurring on-board the lander.

18.3 Baseline Design

The lander module will enclose the TT&C subsystem which is composed of:

Deep Space transponders, two transponders are considered for redundancy, the transponder receivers are working in hot redundancy while the transmitters will be working in cold redundancy. The transponder will implement ranging and DOR capabilities as requested in REQ LM-250.

The transponder will also need to implement tones MFSK for transmission during the landing phase.

Turbo codes with rate ½ and ¼ with a frame length of 8920 are baselined. The Turbo code ½ will be used when the distance to Earth is small. Turbo codes with rate 1/6 are already available in the NASA Deep Space Network and will be available at the ESA Deep Space Antennas from 2016. It is not known if the Russian ground stations are compatible with this coding scheme.

Modulation: For symbol rates above 60 ksps the use of squarewave subcarrier is not allowed. SP-L or supressed carrier modulation schemes shall then be considered. Suppressed carrier modulation schemes are not compatible with ranging. However the use of filtered SP-L will produce spectral lines that shall be carefully controlled.

Travelling Waveguide Tube Amplifier: two amplifiers are required to ensure the overall subsystem reliability figure. The unit of the Exomars 2016 mission has been selected since it is a fully developed unit. This unit provides 65W of RF output power. It shall be noted that an RF output power of 100 and 120W RF output power are also available.

Each TWTA is composed by a TWT (Travelling Wave Tube) and an EPC (Electronic Power Conditioning).

Radio Frequency Distribution Network, the RFDN provides all connecting elements between the output of the transponder or amplifiers to the antennas.

The RFDN will contain a 3dB coupler to provide the cross-strapping between the transponder transmitter and amplifiers. The diplexer filter will provide the separation between transmit and receive frequencies and provide the filtering to ensure compliance to the emissions and ensure RF auto-compatibility. Waveguides switches and waveguides will also be included to interconnect the transponders/TWTAs to the antennas.

Low Gain Antennas

Antennas implement transmit and receive capabilities. Right hand circular polarisation is baselined.

Three low gain antennas are considered to provide almost omni-directional coverage. These antennas will be used during the LEOP phase, SAFE mode and the entry and landing phases.

The spacecraft configuration makes the location of the antennas on the composite spacecraft difficult. However a working configuration has been found (see system part figures).

IKI suggested that two antennas could be implemented on the propulsion module, however external waveguides will have to be run from the lander to the propulsion module, which will provide additional losses. The antennas will be terminated when the module is separated by a thermal knife and an electrical line shall be provided. This remains as an option that shall be further studied in particular if the coverage analysis shows that the 3 foreseen antennas do not provide full coverage.

The LGA baselined are based on the Herschel Planck , GAIA and Exomars antennas.

High Gain Antenna

A 1.4-meter antenna is considered. The high gain antenna will need very accurate pointing, a pointing accuracy of 0.2 deg it is assumed for the link budget calculations. A two-axis steerable antenna is considered.

18.4 Link Budget

The High Gain antenna shall guarantee a transmit gain around 41 dBi within a pointing accuracy of +/-0.2 deg.

The New Norcia ground station is considered as baseline; the G/T @ 10 degrees elevation angle is 49.2 dBK for a 95% weather availability.

The output power is provided by the TWTA, 65 W are assumed.

Waveguide losses are 0.1 dB/m, bending and flexible waveguides will cause additional losses, switch insertion loss: 0.05 dB, isolator 0.15,

A data rate of 32 kbps is sufficient to transmit the data generated in the different phases except for the sampling site imaging phase. During this phase up to 3 Gbits could be transmitted and therefore during this specific phase it is recommended that the ground coverage is extended, otherwise 14 Phobos days will be needed. However the critical information to be downloaded in real time should be lower than 3 Gbits (in particular the sampling area image(s) that is mandatory to send to the ground just after landing for the sampling point selection, is expected to be quite less than 1 Gbit, This is to be further consolidated in future phases).

18.5 List of Equipment

Table 18-1 and Table 18-2 show the mass and power budgets respectively.

Element 1	Lander			MASS [kg]				
Unit	Unit Name Click on button above to insert new unit	Part of custom subsystem	Quantity	Mass per quantity excl. margin	Maturity Level	cell name	Margin	Total Mass incl. margin
1	Transponder		2.00	3.50	Fully developed	e1_unit1_margin	5	7.4
2	LGA		3.00	0.30	Fully developed	e1_unit2_margin	5	0.9
3	TWTA		2.00	2.10	Fully developed	e1_unit3_margin	5	4.4
4	RFDN		1.00	3.00	To be modified	e1_unit4_margin	10	3.3
5	HGA		1.00	10.00	To be modified	e1_unit5_margin	10	11.0
-						Do not use		
	SUBSYSTEM TOTAL		5	25.1		e1_ss_tot_margin	7.6	27.0

Table 18-1: Equipment list and mass budget

		Mode		
Equipment	**Number of units active**	**Reception**	**Reception &Transmission**	**Reception &Stand-By Tx**
TRSP – Rx	2	18 W	18 W	14 W
TRSP – Tx	1	0 W	38 W	38 W
TWTA	1	0 W	100 W	13.5 W
Total		36 W	174 W	79.5 W

Table 18-2: Power budget

Note: The receivers are operated in hot redundancy while the transmitters and amplifiers are operated in cold redundancy.

Stand-by mode: The TWTA is in pre-operational mode; the high voltage is OFF when there is no transmission.

18.6 Options

Options have been identified in the previous sections.

18.7 Technology Requirements

No new technologies have been identified for this domain:

However the following points shall be closely followed:

- Deployment of the Turbo codes rate 1/6 in the ESA ground stations
- Russian ground station performances
- LGA gain patterns on the spacecraft.
- Possibly wave guide connector between the LM and the PM

19 ERC THERMAL PROTECTION SYSTEM

Scope of this chapter is to describe the ERC thermal protection design.

19.1 Assumptions and Trade-Offs

This study was performed after two other CDF studies (MMSR and MMSR-A5), as well as two industrial studies (Phootprint pre-phase A's) both assessing the feasibility and the preliminary design of a Phobos Sample Return mission.

Most of the material presented in the following pages has been directly extracted from the reports of the previous activities.

Hereafter a series of assumptions are listed which have driven the design of the ERC from a thermal point of view:

It has been decided to use a European low density TPS material called ASTERM (similar to the American PICA) which is currently under development for the front shield and a second material already space qualified called Norcoat Liege for the back cover.

Norcoat Liege, a cork based material with a density of 470 kg/m^3, is more suited for the range of heat fluxes acting on the back cover. It can sustain fluxes up to 2 MW/m^2.

ASTERM, a carbon phenolic material, has a density of 280 kg/m^3 (denser versions, 350 and ~500 kg/m^3, can also be produced) and was successfully tested for a combination of heat flux and stagnation pressure as follows:

- Peak heat flux 15-16 MW/m^2
- Peak stagnation pressure 0.8 ÷ 1 atm.

This combination of heat flux and stagnation pressure has to be understood as boundary line. Low density carbon phenolic materials cannot be used anymore for higher values of peak heat flux and stagnation pressure. Denser, thus much heavier, materials have to be considered in those cases (densities in the order of 1400-1600 kg/m^3).

The ERC geometry has been based on the Hayabusa one as this shape has shown to be the most stable (without a parachute) within the full flight regimes (supersonic, transonic, subsonic).

A series of margins have been applied during the TPS computation:

- Margins on the Aerothermodynamics fluxes:
  - Convective fluxes: +20%
  - Radiative fluxes: +100%
- Margin on the cold structure max allowable temperature = -10 C (cold structure designed at 160 C but can sustain 170-180 C) (predicted vs. calculated values)
- Margin on the computed TPS thickness: +10% (uncertainty)
- Margin on overall TPS mass: +20% (but 0% at system level) (maturity).

The TPS thicknesses have been calculated in such a way to guarantee that the cold structure maximum allowable temperature of 160 C is met up to hard crash landing and that the sample container maximum temperature does not exceed 40 C up to recovery (assumed occurring within 4 hours after landing).

In the following picture and table the total heat flux (sum of convective and radiative fluxes including margins) is shown for two different trajectories:

- Max heat flux trajectory considered for TPS Ablator selection
- Max heat load trajectory considered for TPS thickness computation

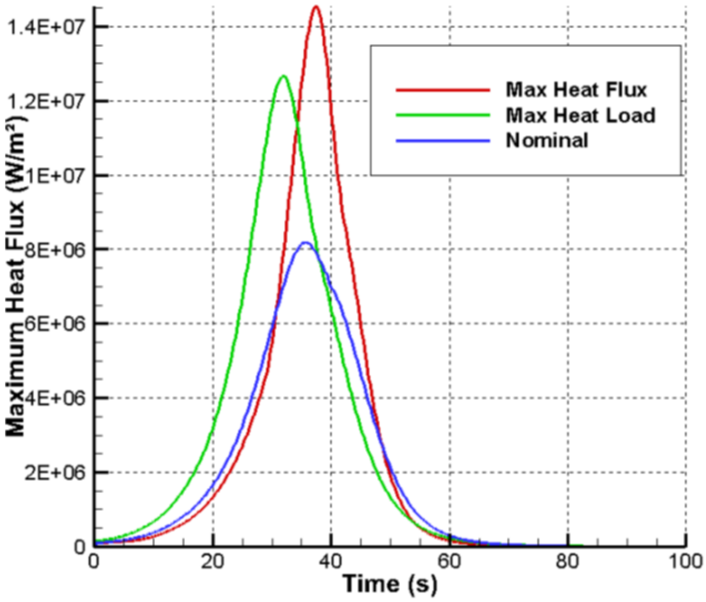

Figure 19-1: Tot heat fluxes for nominal, max heat flux, max heat load trajectories

Trajectory	Max. Heat Flux	Nominal	Max. Heat Load
FPA	Maximum (Steep)	Nominal	Minimum (Shallow)
FPA (deg)	-10	-9.8	-9.6
Maximum Heat Flux (MW/m^2)	15	8.2	12.7
Maximum Heat Load (MJ/m^2)	221.3	178.8	235.8
Total Pressure at peak heat flux (kPa)	36.4	25.3	30.1
Maximum deceleration during entry (g)	60.5	42.2	44.0
Impact velocity (m/s)	41	38	37

Table 19-1: Trajectory principal characteristics

Note: The scaling factor to assess the total heat flux to be used on the back cover is in the range 1-10% (with respect to the front shield stagnation point total heat flux).

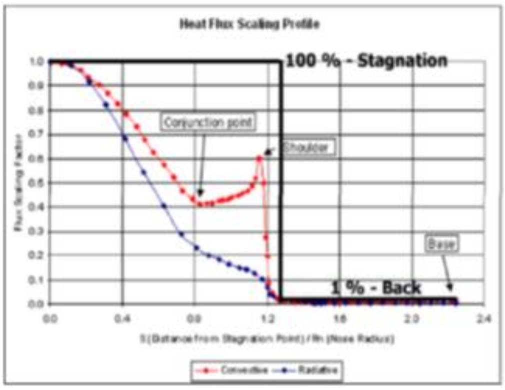

Figure 19-2: Heat flux evolution between front shield and back cover

A further assumption is that the ERC and the other mission elements are thermally decoupled.

19.2 Baseline Design

With the assumption described in the previous chapter the overall TPS design has given thicknesses of 60 mm and 11 mm (20 mm locally on the lid area) for the front shield and the back cover respectively. The overall mass is about 13.5 kg including maturity margin.

Element 2	Earth Re-entry Capsule			MASS [kg]			
Unit	Unit Name Click on button above to insert new unit	Part of subsystem	Quantity	Mass per quantity excl. margin	Maturity Level	Margin	Total Mass incl. margin
1	FS TPS (60mm ASTEM material)		1	8.98	To be developed	20	10.78
2	BC TPS (11mm Norcoat Liege (20mm locally on Lid))		1	2.20	To be developed	20	2.64
3					To be developed	20	0.0
-	Click on button below to insert new unit			0.0	To be developed	20	0.0
	SUBSYSTEM TOTAL		2	11.18		20.0	13.42

Table 19-2: TPS mass budget

19.3 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Included in this table are:

- Technologies to be (further) developed
- Technologies available within European non-space sector(s)
- Technologies identified as coming from outside ESA member states.

Equipment and Text Reference	Technology	Suppliers and TRL	Technology from Non-Space Sectors	Additional Information
ASTERM	TPS material	ASTRIUM – TRL5		
Norcoat Liege	TPS material	ASTRIUM – TRL9		

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20 THERMAL

The purpose of the thermal control system is to maintain acceptable spacecraft temperatures throughout the multiple environments and operational phases of the mission. Specific analysis has been performed to assess Phobos Sample Return baseline design.

20.1 Requirements and Design Drivers

From the thermal point of view, the objective of this study has been focused on assessing the feasibility of a thermal control system that would fulfil the temperature requirements of the different equipment of the lander module.

The unit operating and non-operating temperature limits are presented in Table 20-1.

Subsystem	Unit	Temperature Limits [°C]		
		$T_{OP,max}$	$T_{OP,min}$	$T_{NOP,min}$
Avionics	OBC	40	-20	-45
	MMU	40	-20	-45
	PCDU	40	-20	N/A
	Battery	20	0	N/A
	Reaction Wheels	70	-30	N/A
	TWTA	70	-20	N/A
	RX	40	-20	N/A
	TX	40	-20	N/A
	Radar Altimeter	40	-20	N/A
	IMU	40	-20	N/A
Surface Instruments	Stereo Cam	50	-55	-55
	CLUPI	50	-20	-30
	ADRON-RM	40	-20	-30
	DIAMOND	40	-20	-50
	LIBRATION	40	-20	-50
Orbit Instruments	IME	50	-50	-55
	VISNIR	50	-20	-30
	MIDIR	15	5	-40
	WAC	50	-55	-55
	NAC	50	-50	-55
	CSU	50	-55	-55

Table 20-1: Unit temperature limits for each subsystem

It is to be noted that a few of these requirements have been assumed based on similarity from other CDF studies (mainly for the avionics) while the temperature requirements for the instruments were provided in the course of the study. No particular temperature gradient requirements were given to the thermal subsystem, and therefore no specific assessment was carried out on this aspect.

In order to control the temperatures within their limits, passive control is first considered including the selection of thermal coatings, insulation and total radiator area. After the required radiator area is determined an estimate on the total active control (heating power) is also made.

In this study only external radiative couplings are considered, with each internal unit assumed to deliver its heat dissipation to an external radiator. An estimation of Sun and planetary fluxes is also made during three different mission phases: cruise, Phobos orbit and surface operation.

20.2 Assumptions and Trade-Offs

20.2.1 Thermal Assumptions

From a thermal perspective three distinct environments along the mission have been assumed:

1. Cruise from Earth to Mars orbit
2. Phobos orbit
3. Phobos surface

Table 20-2 shows the range of thermal environments encountered whilst operating in proximity of Mars and Phobos.

Phobos IR temp. range	109 – 305 K
Mars IR temp. range	130 – 308 K
Phobos albedo	0.065 - 0.090
Mars albedo	0.17 - 0.25
Mars Solar declination	+ / - 25.2°

Table 20-2: Thermal environment around Mars and Phobos

Additionally the following thermo-optical properties have been considered for the thermal finishes:

Type	BOL	
	α	ε
External MLI	0.56	0.75
SSM	0.12	0.75
Black Paint	0.98	0.91
Solar Cells	0.92	0.80

Table 20-3: Thermo-optical properties

Table 20-4 summarises the dissipation figures. When no values are provided for the stand-by, the unit is assumed to be always ON with only 80% of the total dissipation (to simulate cold conditions).

Subsystem	Unit	Dissipation [W]	
		ON	STDBY
Avionics	OBC	30	
	MMU	30	
	PCDU	70	
	Battery	15	
	Reaction Wheels	20	
	TWTA	35	20
	RX	28	
	TX	38	
	Radar Altimeter	8	
	IMU	12	
Surface Instruments	Stereo Cam	11.5	8
	CLUPI	12.5	3.3
	ADRON-RM	5	5
	DIAMOND	10	10
	LIBRATION	10	10
Orbit Instruments	IME	5	5
	VISNIR	20	7
	MIDIR	18	4
	WAC	11.5	8
	NAC	15.6	8
	CSU	5	5

Table 20-4: Internal unit dissipations

It is to be noted that the surface instrument are considered always OFF except in surface operation while the orbit instruments are assumed ON only while orbiting Phobos. The avionics are considered always ON while in cruise and in orbit. Only the reaction wheels, the IMU and the radar altimeter were assumed OFF during the surface operations. A summary of the units activation is provided in Table 20-5 below.

Phase	Case	Instruments		
		Avionics	Orbit	Surface
Cruise	**Hot**	ON	OFF	OFF
	Cold	ON	OFF	OFF
Orbit	**Hot**	ON	ON	OFF

Phase	Case	Instruments		
		Avionics	Orbit	Surface
	Cold	80%	OFF	OFF
Surface	Hot	ON	OFF	ON
	Cold	80%	OFF	OFF

Table 20-5: Dissipation modes of the subsystems for each mission phase

20.2.1.1 Cruise from Earth to Mars

During the first phase of the mission the solar constants chosen for analysis are 1412 and 490 W/m² resembling a winter solstice (WS) departure from Earth and a summer solstice (SS) arrival at Mars. Only the Sun flux is considered during the interplanetary transfer with rays arriving on the solar array at 90° angle of incidence. Therefore the variation in solar constant magnitude defines the hot and cold cases for the steady state.

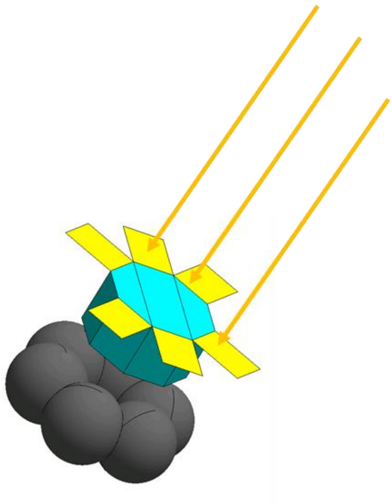


Figure 20-1: S/C attitude during cruise

20.2.1.2 Phobos Orbit

To define the thermal environment during this mission phase, an average orbit at 70km altitude around Phobos is considered with fluxes from the Sun, Mars and Phobos itself contributing to the heating of the spacecraft. The hot and cold cases are sized at WS and SS respectively over the range of thermal properties shown in Table 20-2. An average of the fluxes has been made in order to perform a steady state analysis.

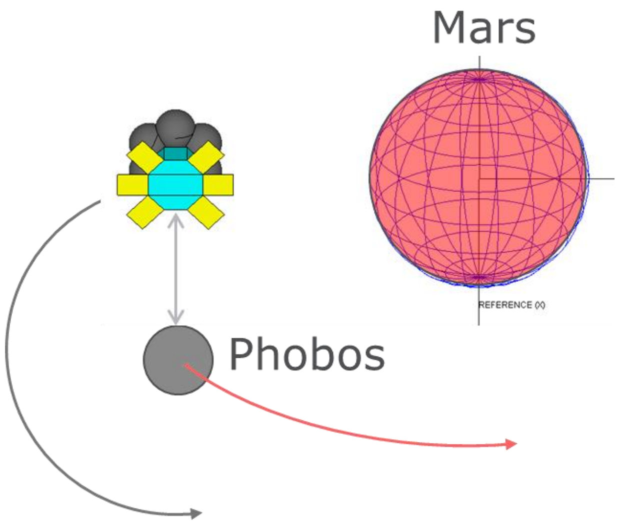


Figure 20-2: S/C attitude during orbit around Phobos

20.2.1.3 Phobos Surface

Surface operation are constrained to be occurring during Mars winter solstice for power management purposes. The hot case has therefore been defined from landing on the south pole of Phobos during winter solstice while the cold case, is sized from a landing at -25° latitude on Phobos with no view factor to Mars at the same season. Note that the latitude constraint was put in place at the time of the analysis for power reasons to ensure a balanced day/night duration (50-50%).

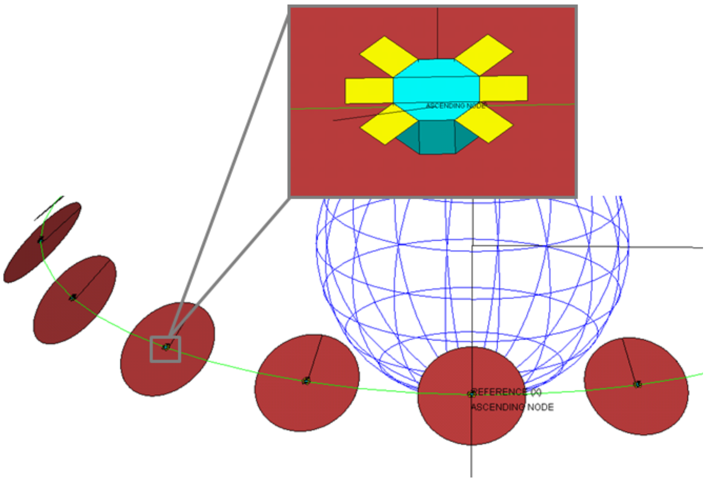


Figure 20-3: S/C attitude during surface operation

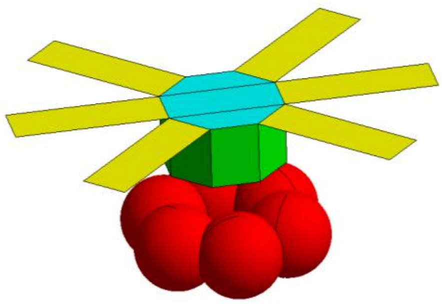
20.2.2 ESATAN-TMS Thermal Model

A Geometrical Mathematical Model has been developed in order to determine the view factors between surfaces along with the impinging heat fluxes from various sources. It is to be noted that in the surface case, the view factors to deep space are greatly reduced by a large view factor to Phobos surface.

The landing module is comprised of an octagonal structure with inactive interfaces to the PM and ERV. The eight rectangular side panels are considered for radiator placement including six with a large view factor to the rear of the solar array. The solar array is positioned on the six sides perpendicular to the lander side panels. Note that during the study the LM design has evolved towards a 5 solar panels configuration, but the thermal analyses were not updated. It is expected that conclusions would not vary much.

The PM model is based on the six tank Fregat module.

Figure 20-4 shows the simplified model of the spacecraft. Internal heat transfer within the lander module is not considered. It is therefore assumed that the internal units are arranged to dissipate their heat through the lander side panel radiators.



	Solar cells
	SSM
	MLI
	Black paint (reverse solar array) – not shown
	Inactive

Figure 20-4: Lander and PM thermo-optical properties

It should be noted that the PM and ERV interfaces, lander legs, robotic arm and antennas are not considered in this model.

20.2.3 Results

Since the definition of the configuration was floating until late during the study, a worst case scenario has been selected for the sizing of the radiator and the heater power.

Therefore the following method has been used:

Radiator Sizing

1. Assume each unit individually (except of the OBC+MMU)
2. Assume a target radiator temperature at T_{max} - 5K for radiator sizing
3. Determine the most illuminated lander panel along with lowest field of view to space
4. Size radiator for this unit with selected environment
5. Repeat the exercise for all mission phases i.e. cruising, Phobos orbit, surface operation
6. Finalise the radiator size by selecting the most demanding case

Heater Sizing

7. Assume a target radiator temperature at T_{min} + 5K
8. Determine the least illuminated lander panel along with the highest field of view to space
9. Size heater power for this unit with selected environment for each mission phases

This methodology presents the advantage of covering all possible cases of S/C attitude during landing and orbit around Phobos. However a limitation appears for items with very stringent temperature requirements such as the battery, for which a more precise knowledge of the panel illumination is needed, with potential constraints on the attitude control. This will undoubtedly be further iterated in the next phase of the project.

	Radiator Area [m²]	Sizing Case	Cruise Power [W]	Orbit Power [W]	Surface Power [W]	Remark
OBC + MMU	0.5537	Surface	6	8	0	POW with 80% Dissipation
Battery	1.3337		208	214	157	
PCDU	0.6460		8	10	0	
Reaction Wheels	0.0487	Orbit	0	0	0	Surface POW: Not used
Transponder	0.3407	Surface	4	5	0	POW with 20% Dissipation
IMU	0.0517	Orbit	0	0	0	Surface POW: INSTR OFF
Radar Altimeter	0.0344		0	0	0	
TRSP RX	0.2584	Surface	0	0	0	
TRSP TX	0.3507		0	0	0	
Stereo Cam	0.0736	Surface	2	2	0	
CLUPI	0.0800		7	7	6	
ADRON-RM	0.0496		4	4	4	
DIAMOND	0.0991		3	4	7	
LIBRATION	0.0991		3	4	7	
VISNIR	0.0773	Orbit	9	10	0	Orbit POW: INSTR OFF Surface POW: Not used
MIDIR	0.1817		37	38	0	
WAC	0.0445		0	1	0	
NAC	0.0603		0	2	0	
CSU	0.0193		0	1	0	
IME	0.0193		0	1	0	
Total + 20% Margin:	**5.3**		**349**	**373**	**217**	

Table 20-6: Results with selected methodology

As can be seen in Table 20-6 the batteries are driving the total radiator size and the heater power. A simple optimisation of the battery location could significantly reduce the radiator area needed. Let's assume for example that the attitude of the spacecraft is such that when landed the battery is located on the least illuminated face, however still under a solar array for the radiator sizing. The coldest face can be kept for the heater power sizing. The new results would then be as depicted in Table 20-7.

	Radiator Area [m²]	Sizing Case	Cruise Power [W]	Orbit Power [W]	Surface Power [W]	Remark
OBC + MMU	0.5537	Surface	6	8	0	POW with 80% Dissipation
Battery	0.4119		56	58	41	
PCDU	0.6460		8	10	0	
Reaction Wheels	0.0487	Orbit	0	0	0	Surface POW: Not used
Transponder	0.3407	Surface	4	5	0	POW with 20% Dissipation
IMU	0.0517	Orbit	0	0	0	Surface POW: INSTR OFF
Radar Altimeter	0.0344		0	0	0	
TRSP RX	0.2584	Surface	0	0	0	
TRSP TX	0.3507		0	0	0	
Stereo Cam	0.0736	Surface	2	2	0	
CLUPI	0.0800		7	7	6	
ADRON-RM	0.0496		4	4	4	
DIAMOND	0.0991		3	4	7	
LIBRATION	0.0991		3	4	7	
VISNIR	0.0773	Orbit	9	10	0	Orbit POW: INSTR OFF Surface POW: Not used
MIDIR	0.1817		37	38	0	
WAC	0.0445		0	1	0	
NAC	0.0603		0	2	0	
CSU	0.0193		0	1	0	
IME	0.0193		0	1	0	
Total + 20% Margin:	**4.2**		**167**	**186**	**78**	

Table 20-7: Results with optimised battery location

The results become suddenly more acceptable, highlighting the need of a more consolidated arrangement of the units, and of considering constraining the spacecraft attitude at landing. This exercise is unfortunately outside the scope of this study and should be performed in a next phase.

20.3 Baseline Design

The selected design is making use of well known hardware, such as second surface mirror tapes for the radiators, heaters with temperature sensors and MLI for the surfaces not used as radiators. Although not mentioned, it is expected to black paint most of the internal parts of the lander.

20.3.1 Budgets

With the selected baseline design the power requirements for the TCS sub-system could amount to the figures presented in Table 20-6 or Table 20-7 depending on the constraints to attitude control that the TCS could impose.

Mass wise the design selected is particularly light and should not exceed 12 kg including 20% margins.

21 GS&OPS

21.1 Requirements and Design Drivers

The mission is a direct cruise to Mars with or without DSMs (in addition to the Launcher Insertion Correction and its subsequent correction manoeuvre). Following the QSO at Deimos and the short transfer to Phobos, the PM is jettisoned into a benign orbit and is no longer a consideration for operations. Following a Trailing Orbit phase and a QSO phase, a minimum of 3 fly-bys are required over the selected landing site before the final autonomous landing is commanded. Sampling and ERV ascent operations occur within one Earth week. The LM remains on the surface for a TBD period of science operations.

On the return cruise the ERC is separated a few hours prior to Earth arrival for a passive hyperbolic re-entry into the atmosphere. The landing is without parachutes and the samples are protected by compressible foam. The ERV is put on an Earth avoidance course and disposed of.

Between the mission sub-options of the System baseline design, there is no difference identified in the expected mission events, other than in the duration of the QSO phases around Deimos and Phobos, and the trailing orbit ahead of Phobos. The Long Mission Scenario has the disadvantage of a much longer lifetime (5y vs 3y for the Short Mission Scenario), but its timeline of operations around the Martian satellites is much more relaxed with the related benefits that can be expected for the management of the on-ground resources (in addition to the added science possible).

The mission has obvious similarities with previous studies such as Phootprint but with the new element of cooperation with ROSCOSMOS. A table of baseline responsibility sharing is available in the Systems section but can be summarised as ESA Operations from launcher separation (i.e. LEOP) up to the end of Surface Operations, including sampling but excluding the sample return, and ROS Operations from ERV launch up to ERC separation and ERV disposal. The capsule retrieval operations described in RD[29] are considered still relevant and not discussed further here.

MI-160 requires that the mission design shall be such that operations can take place from a single ground station during non-critical phases whilst for critical phases, the need of more than one ground station shall be justified. Aside from defining what is critical, it should be noted that navigation needs will require the use of more than one ground station (e.g. for ΔDOR and for the elimination of systematic errors for Mars approach) and that timing issues at Mars will require the use of a specific station from one of a set of qualified deep space stations (e.g. the deep space stations of the ESTRACK and ROS networks). In addition, following ERV launch, there will be two elements with need of communications back to Earth: the ERV and the LM.

21.2 Assumptions and Trade-Offs

A mission phase is understood as being critical, if a failure or an underperformance can cause mission loss or permanent degradation, i.e. LEOP, Mars insertion manoeuvre, manoeuvres in Phobos/Deimos proximity, descent/landing/surface sampling operations/ascent, the Mars escape sequence and ERC separation.

The spacecraft will spend most of its time at distances greater than 0.1AU from the Earth where the one-way light-time delay in communications is already in the order of one minute. This requires the operational mission to be considered as “off-line” for which it is assumed that the on-board systems will be robust and have an advanced level of autonomy.

It is assumed that there shall be a sharing of ground station resources between ESA and ROS on a zero-exchange of funds basis.

The Phobos and Deimos QSO phases are assumed to be operationally identical.

An accurate ground-based orbit determination campaign for the science orbits is performed using both Doppler/Ranging and relative measurements from the spacecraft to the surface (i.e. by camera and altimeter) possibly with landmark determination. Relative knowledge within a few meters is expected. For manoeuvres there is then a measurements data cut-off point from when the ground computes and checks the manoeuvres profile and finally uploads it (and during which the relative knowledge error increases). This method applies up to the starting point of the Phobos descent when autonomous on-board navigation takes over (which has to be verified under representative conditions).

The Phobos day lasts for approximately 7h 39m with approximately 2hrs Earth contact time available per Phobos day from the surface, i.e. three 2h passes per Earth day.

Ground communications during the landing are only available via the LGAs.

It is assumed that a mock-up or engineering model of the robotic devices (arm and sample transfer/sealing mechanisms) can be made available to the MOC for training/validation and for troubleshooting purposes (e.g. EM or ETB kept operational at the Prime’s premises), and that the PIs provide support for their respective instruments after launch.

It is assumed that the on surface commissioning operations of the robotic devices and the payload can be performed in parallel to the selection of the sampling site.

It is assumed that the mission will include one commissioning phase of a few months immediately after launch, and another one prior to Mars orbit injection and/or descent /landing.

It is assumed that for all mission phases besides descent/landing the spacecraft is capable of downlinking the telemetry in the allocated tracking time.

During critical phases, in particular the Phobos descent/landing, at least a minimum set of essential HKTM are provided to allow for post factum determination of major possible causes in case of failure.

It is assumed that the spacecraft can be operated via a timeline, including in particular autonomous slews, changing of instruments, initialisation of instruments, and thermal control settings, and that it is sufficient that Mission Planning will be supported during normal working hours of the Flight Control Team only. The exception to this is for critical operations as defined above where LEOP-like conditions will be in effect.

The master timeline shall be able to cover up to 7 days of nominal operations. During interplanetary transfer, the master timeline shall be able to cover at least 2 weeks of operations.

It is assumed that all forms of space-ground communications and data encoding are compliant with the standards in use at the Mission Operations Centre.

It is assumed that there are no science operations during the cruise phases to and from Mars.

21.3 Baseline Design

There shall be a single ESA MOC for the PM/LM composite (including sampling operations) and a ROS MOC for the ERV/ERC ascent and return operations with an unrestricted bi-directional exchange of data via their respective ground segments. In contrast, there shall be a single Science Operations Centre for the entire mission.

Nominal spacecraft control during most of the commissioning, cruise, and Mars phases shall be "off-line". Only one ground station will be allocated for communications with the spacecraft during these phases, except for critical events. Dual ground station coverage will be used when required for navigation during cruise and for limited special operations during the Mars phase. During cruise the nominal coverage will be limited to a single pass per fortnight. That implies that the Phobos-SR spacecraft is assumed to provide on-board capabilities such that the satellite is able to perform corrective actions in case of on-board anomalies and the ground segment does not need to monitor the spacecraft in real time.

Phase	**Duration (d)**	**Comment**
LEOP	3	• Close to 24h coverage by 3 DSAs, • continuous on-console FCT support, • high redundancy of services, • real-time expert support
Commissioning	60	• Single station coverage, 10hrs / day, • mainly platform ops, • limited payload ops until at Deimos or on Phobos surface
Cruise outbound	202	• Five 8h passes / week for the first half of the phase to build-up confidence/ experience in the spacecraft, • reducing to one 8-10h pass per fortnight in the second half, • FCT preparations of Mars Ops continue
Mars approach and orbit insertion	90	• Increasing coverage including Delta DOR for ground-based radiometric navigation, • FCT simulations and dress rehearsals of orbit insertion and TOAs • Dual station coverage for the insertion

Phase	Duration (d)	Comment
Target Orbit Acquisition (Deimos)	30	• Ongoing tracking campaign with daily range/Doppler plus Delta DOR as required • Manoeuvres commanded from ground • Dual station coverage for the burns
QSO around Deimos	29/304	• Single station coverage, 8hrs/day • Routine, office-hours only FCT support • Preparations for transfer to Phobos
Transfer to Phobos orbit	1	• Dual station coverage for the 6hrs of the Hohmann transfer • On-console FCT support
Trailing Orbit ahead of Phobos	20/100	• Single station coverage, 8hrs/day • Routine, office-hours only FCT support • Relative navigation measurements driving the orbit maintenance profile • Science Ops to increase the knowledge of ephemerides, gravity field, surface contours, and landmarks to ensure safe fly-bys
QSO around Phobos	100/365	
Fly-bys over selected landing site	25	• 11hrs/day required by TT&C
Landing preparations	12	• Single station coverage, 8hrs/day • Time for ground staff to implement updates, corrections, etc. based on experience from the fly-bys • Dress rehearsal
Landing operations	2	• Operations as for LEOP with shift operations and block booking of supporting deep space stations during Mars visibility • Once on the surface, at least single station coverage, 3x2hrs/day • All steps are preplanned and final upon the GO decision
Surface operations: sample point selection	3	
Surface operations: sample acquisition & transfer	3	
Surface operations: in-situ observations	180 (TBC)	• ESA operations that continue in parallel to the ascent and Earth return operations to be executed by ROS • Single station coverage, 3x2hrs/day or as required for science data return
Ascent	2	• ROS operations • Dual station coverage with ESA • ESA maintains focus on LM
Departure orbit acquisition	28	• ROS operations

Phase	Duration (d)	Comment
Trans Earth injection	2	• Single station coverage, 8hrs/day
Cruise to Earth	270	• Five 8h passes / week for the first half of the phase to build-up confidence/ experience in the spacecraft, • reducing to one 8-10h pass per fortnight in the second half
Earth Approach	30	• Single station coverage, 8hrs/day
ERC targeting/release and ERV disposal	10	• Dual station coverage for the one or more targeting manoeuvres, then the ERC release and ERV avoidance manoeuvre at re-entry time minus 4hrs • Then one station per element

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22 PROGRAMMATICS

22.1 Requirements and Design Drivers

The main requirements and design drivers for the Phobos Sample Return mission from a programmatics point of view are:

- Launch: 2024, backup 2026
- A satellite with four modules (Propulsion Module, Lander Module, Earth Return Vehicle, Earth Return Capsule)
- Cooperation between ESA and ROS/IKI/LAV
- Equipment and applied technologies shall have reached at least TRL 5 at the start of the mission implementation phase.

22.2 Assumptions and Trade-Offs

The main assumptions for programmatics are:

- The PM and the ERV are under responsibility of ROSCOSMOS
- The LM and the ERC are under responsibility of ESA.
- As backup solution the responsibility for PM and LM could be swapped (i.e. the PM is under ESA's responsibility and the LM is under responsibility of ROSCOSMOS).
- Launch from Baikonur (Kazakhstan) with Proton and Breeze-M
- ERC Re-entry in Kazakhstan
- The mission Planetary Protection category shall be: category V, unrestricted Earth return (TBC by COSPAR)
- The qualification of all modules is, as far as possible, performed at module level
- PFM models of the modules will be tested at module level for workmanship before integration of the composite satellite for protoflight testing.

Four options were evaluated with different attributions of responsibility (ROSCOSMOS, ESA):

- Option 1: Three modules instead of four. PM and LM are combined to one module
- Option 2: Four modules, but the PM ACS is located on the LM
- Option 3: Four modules, but PM and LM control located on ERV, LM without propulsion subsystem
- Option 4: Each module has its own control system. PM detached before reaching Phobos. For option 4 four variations are investigated with different use of LM and PM propulsion for key manoeuvres.

For the sampling system delivery by either Russia or Europe is considered.

Integration of the four modules and verification at composite level either in Europe or Russia/Kazakhstan is evaluated.

22.3 Model Philosophy

A Hybrid Model Philosophy (RD[32]) is the baseline approach for this project, with SM, AVM and PFM. Additional models will be introduced as needed e.g. a scaled model for wind tunnel tests of the ERC, models for the landing mechanism and for the sampling mechanism.

Assuming that the PM and ERV are provided by ROSCOSMOS, this chapter (written by ESA) concentrates on the models needed for the LM, ERC and subsystem of them.

The four options lead to different complexity attribution of PM, LM and ERV. For the LM, the only European of these modules, the impact on the various models is described in Table 22-1.

A simulator will be needed to develop and validate proximity and landing operations. The simulation results will be used as input to the Lander GNC PFM testing.

For option 1 a propulsion module qualification model will be needed for thruster firing tests.

	Lander Option 1	Lander Option 2	Lander Option 3	Lander Option 4	ERC	Sampling System	System level verification
Structural Model (SM)	Flight std. Structure, propulsion and mechanism	Flight std. Structure, propulsion and mechanism	Flight std. Structure, and mechanism	Flight std. Structure, propulsion and mechanism	Flight std. structure	Dummy	Mechanical qualification, mechanism functional tests
Scaled Model					Wind tunnel test models (as needed)		
Avionics Verification Model (AVM)	EM active valves and components	Flight std. Structure, propulsion and EBB electronics	Active elements (electrical)	Elegant Breadboard (EBB) units	Functional models of Mechanism	Active elements (electrical)	SW and functional tests
Landing Mechanism (EQM)	Flight std. Structures / Mechanisms	Flight std. Structures / Mechanisms	Flight std. Structures / Mechanisms	Flight std. Structures / Mechanisms	Dummy	Flight std. structures / mechanisms	
Impact (Earth) test models					Flight std. structure and sample locking mechanism		
Sampling mechanism qualification model	Partial flight std. structure and QM mechanism	Partial flight std. structure and QM mechanism	Partial flight std. structure and QM mechanism	Partial flight std. structure and QM mechanism	Partial flight std. model (ERC QM mechanisms)	Flight std. and complete	Tests at mechanism ERC and system level
Proto-Flight Model (PFM)	Refurbished SM + PFM Avionics + TCS	Refurbished SM + PFM Avionics + TCS	Refurbished SM + TCS	Refurbished SM + PFM Avionics + TCS	Full flight std.	Full flight std.	Proto-qualification tests, functional tests, mass properties measurements
GNC validation models	GNC Elegant Breadboard (EBB) electronics, sensors	GNC EBB electronics, sensors		GNC EBB electronics, sensors			
Responsibility	Europe	Europe	Europe	Europe	Europe	Europe and/or Russia?	Europe or Russia?

Table 22-1: The model philosophy

22.3.1 Landing Module

The Landing Module will be a new design with a medium size (wet mass > 1000kg). During launch it will sit on the large propulsion module (wet mass > 3000 kg) and it will carry the ERV (wet mass > 450 kg) and ERC (m < 50 kg).

Because of the complex satellite configuration with modules with different heritage, a structural model of the LM is needed for qualification at module level.

No PM SM is needed if the PM interface is simulated by an adapter with equivalent stiffness.

The ERV and ERC can be replaced by dummy structures if analysis confirms that they do not influence the dynamic behaviour of the composite, otherwise they need to be simulated by dynamically representative structures or simply by SM models.

Preliminarily it is assumed that the thermal design is rather simple and thermal qualification of the LM can be postponed to the composite PFM tests.

For cost saving it is anticipated that the LM SM can be refurbished and re-used for the PFM.

Note that Lavochkin indicated that the separation test of the LM using the Russian PM-LM interface adaptor should be performed in Russia: this means that the LM SM should be transported to Russia for this test.

An AVM is proposed for the Lander, allowing its functional qualification testing and integrated functional tests with the spacecraft. Equipment Engineering Models will be needed for Option 1, 2 and 4 while for Option 3 “Elegant Breadboard” (EBB) units might be sufficient. The procurement of EBB or EM units would be from the PFM unit suppliers.

For the specific Mars moon landing features of this mission, qualification is needed (touch-down loads, proximity operations). As a consequence of this, the following elements will need qualification models: landing legs, sampling system, including a dedicated qualification model for the sampling mechanism itself.

For all four options integrated functional test will have to include approach and landing operations too. Design and simulation of proximity operations may benefit from existing analyses and experience of planetary or small body operations, e.g. from the ESA mission Rosetta.

22.3.2 Sampling System

A qualification model of the sampling system is considered necessary (including ERC part). All parts of the sampling system need to be at TRL ≥ 5 before the start of the Implementation Phase i.e. an EBB or EM of the whole sampling system shall have been successfully validated before start of the implementation phase, preferably, already by the SRR.

The test of sampling has to be designed to demonstrate proper implementation of the required sampling function. The testability of such a system for acceptance is to be assessed as part of the design.

22.3.3 Earth Re-entry Capsule ERC

The ERC will be mostly passive except for its landing beacon and its sample closure mechanism. Its heat shield will be designed and manufactured from qualified TPS material. ESA is testing such material and qualification of it is expected to be achieved by PDR.

No parachute will support the last phase of descent and landing therefore impact test models of the whole capsule are necessary (Structural Model – SM), for test verification of landing impact, recovery operations and sample containment integrity.

A first estimation of the needed quantity "n" of impact tests is n≥ 2. Scaled models for wind tunnel testing (low and high speeds) will be used to confirm the design choices for the ERC aerodynamics and ballistic coefficient.

A model for qualification of the sample containment mechanism by test is needed (QM).

22.3.4 Earth Return Vehicle ERV and PM

PM and ERV are assumed to be delivered by Russia. This implies the delivery of FM or PFM for integration of the satellite composite for PFM testing.

Before that AVM or reduced AVM of PM and ERV, depending on the option, will be needed for combination with the LM AVM and ERC AVM functional testing.

SM of PM and ERV might be useful for high fidelity composite SM testing, but might be replaced by representative structures as described above.

22.4 Technology Readiness

Table 22-2 identifies equipment envisaged for use on this mission with a TRL of 5 or lower. The source of this information is largely coming from the Phootprint project. A TRL of 2, which has been identified in a few cases, requires typically about 6 years development funded by the TRP programme before reaching TRL 5 and then another 4 years development funded by the project before being ready for integration on a satellite. Therefore a technology development plan is needed for all equipment and software which is very low.

Module	**Item**	**TRL**
ERC	Internal Structure	5
ERC	Energy Absorbing Material (Aluminium foam) 1)	3-4
ERC	Canister Interface	5
ERC	Beacon System	2-3
ERC	Spin-up and Eject Mechanism Interface	5
ERC	Front Shield Thermal Protection System	5
ERC	ERC Re-entry simulation and models	4-5
ERC	ERC GSE	2
LM	Inter Module Equipment	5
LM	Deployment Mechanism	4
LM	Image processing / GNC algorithm	4
LM	Landing Legs / damper and deployment mechanism	2-4
LM	LM Li-Ion Battery Module	5
LM	LM Solar Array Assembly	5

LM	HGA APM (Com System)	5
LM	Radio Frequency Distribution + Waveguides (Com System)	4
LM	GNC Wide Angle Camera (AOCS)	4
LM	AOCS / GNC Application SW	2-4
LM	Radar Altimeter (AOCS)	3-4
SATCS	SATCS = Sample Transfer Subsystem	2-5
SATCS	Sample Transfer Arm (4DOF)	4
SATCS	Transfer Arm Electronics	4
SATCS	ERC I/F Mechanism	3
SATCS	Sampling Mechanism	4
SATCS	Sample Container	5
SATCS	Sampling Mechanism Electronics	2
SATCS	Sampling Verification Facility	2

Table 22-2: TRL status

22.5 Verification Approach

The verification approach shall be compliant with RD[34].

As baseline all modules shall be qualified on module level and the module PFM shall be delivered acceptance tested for integration in the satellite composite and the subsequent PFM test campaign.

The PFM test campaign shall be performed in compliance with RD[35] and with launcher requirements, tailored for this project.

The design shall take into account the need of testing at equipment, module and system level for qualification and acceptance or proto-qualification, with the related increased number of load cycles, in particular where structural models are refurbished for PFM use.

Special simulators and tests shall be employed to cover the verification needs which go beyond typical spacecraft testing, e.g. proximity operations, landing, sample collection, sample return.

22.6 Schedule

The schedule of RD[31] has been taken as a reference, but adding a Phase A (12 month), Phase B1 (12 month) and Phase B2 (15 month) upfront. For the Phase CD proposal process (ITT, evaluation, contract negotiation) 9 month are assigned. Note that in case the same would apply in case the ITT is for B2CD phase instead of phase CD (there would be a switch in the planning but the overall duration would be the same).

Some task durations are rounded up, leading to slightly longer durations at module level when compared to RD[31]. The Landing Module and the system level activities are on the critical path as before.

Due to the fact that the Landing Module PFM structure manufacturing is started right away after the SM manufacturing it is on the critical path and not the SM composite tests. The PFM structure manufacturing go-ahead before completing the SM tests is of course associated with risk.

With this assumption a schedule margin (ESA contingency) of 7.5 month is achieved towards a launch date on 20/09/2024.

It should be noted that some not critical tasks in the schedule do not identify their successor. In the Detailed Gantt chart, which visualises the critical path, some tasks (e.g. 10, 13, 39) show therefore slack towards the end of the project.

If the backup solution (i.e. the PM is built by Europe and the LM is built by Russia) is implemented the contingency is likely to be reduced by at least one month. The Phase C/D duration for the concerned modules differs by one month (PM 804 days, LM 825 days), but an additional negative effect comes from the fact that, in contrary to the Russian PM, a European PM will be a completely new development.

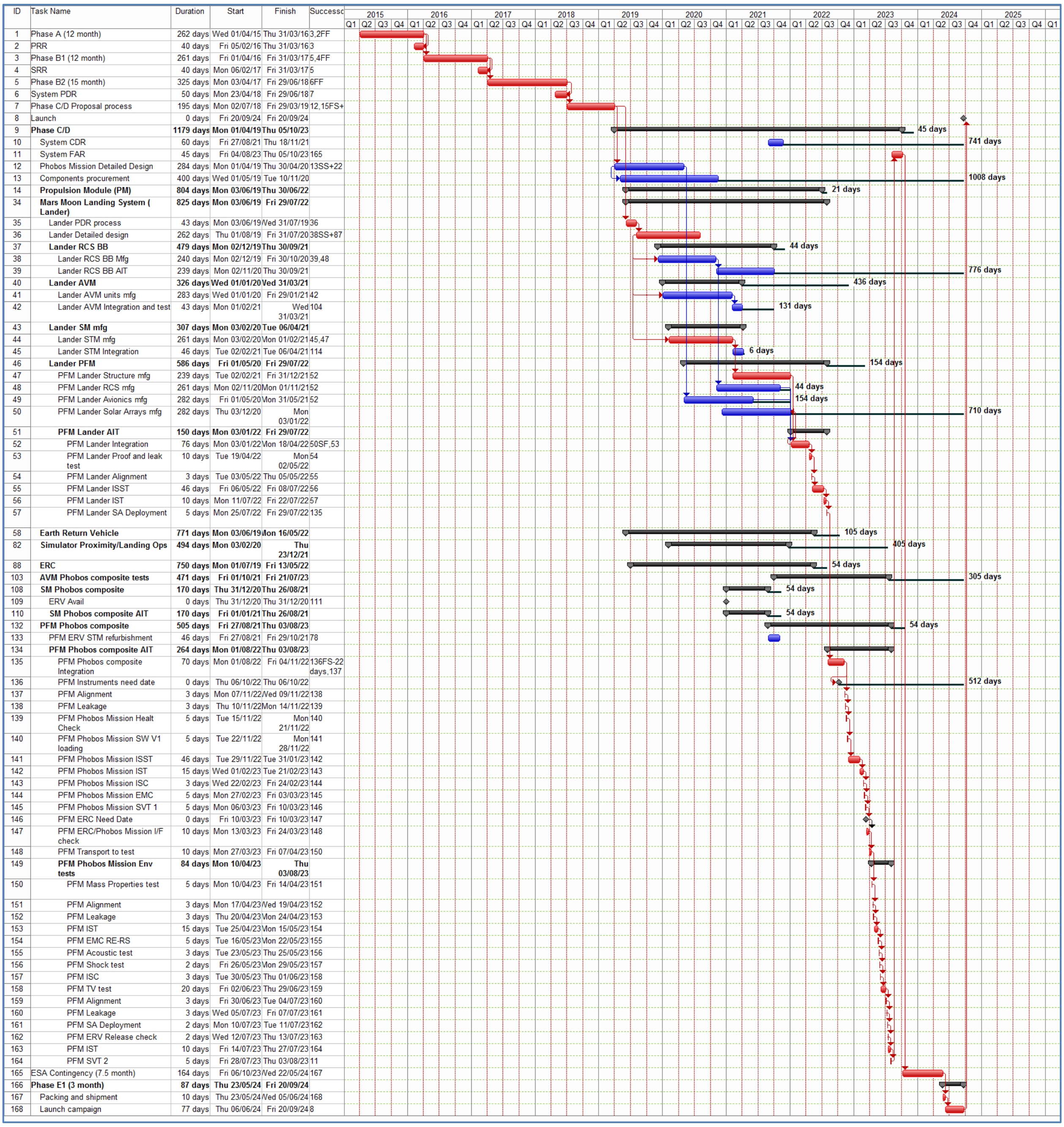


Figure 22-1: Schedule of Option 4 including critical path

22.7 Conclusion and Recommendations

With an envisaged launch date on 20/09/2024 and a Phase A start by April 2015 an ESA contingency of 7.5 months can be accommodated.

This assumes a single break before a Phase C/D implementation phase starting April 2019. For items with very low TRL this means that their development must continue right away as only less than 5 calendar years are left up to that milestone and items with TRL 2 need typically about 6 years to achieve TRL 5. After that about 3 years are left before PFM integration which is less than the 4 years which are typically required.

Accounting for the complexity of the Inter-Agency industrial organisation the break before the implementation phase is proposed before Phase C/D to allow completing of the preliminary design before negotiating the delivery of the complete modules for PM and ERV and to give extra time to advance the development of equipment with low TRL. Nevertheless the durations for Phase A, B1 and B2 are not longer than for other demanding ESA scientific satellites. A move of this break between A/B1 and B2/C/D is possible but comes with disadvantages as explained before. Adding a second break will have a schedule impact of about 6 month.

Also because of the complexity of the Inter-Agency industrial organisation good agreements are needed for who is doing what and the exchange of module internal units has to be avoided. ITAR controlled items shall be avoided as well. The amount of testing at the various levels shall be agreed early in the program taking into consideration that the same items might be tested at several levels (equipment, subsystem, module, composite). The translation of all documents from Russian to English and vice versa is needed as well as joint signature of important documents. An important point is to be careful with the Russian standards which are not disclosed to ESA.

The composite qualification shall be done in Europe, but because handling and testing of the separation mechanism PM/LM needs to be done in Russia, an equivalent structure must be provided for the composite test (could be the LM SM, but impact on schedule would need to be assessed).

23 RISK

23.1 Review of the Mission Requirements Document (MRD)

The mission requirements document was reviewed from a reliability engineering perspective. A change request was suggested for composite requirement number 30 (CO-30) dealing with failure tolerance. The proposed change is described in detail in Table 23-1 below:

	Original	*Change Request*
CO-30	The Composite shall provide single point failure tolerance. Redundancy concepts shall be considered to minimise consequences of single point failures *C: any deviation with respect to this requirement shall be identified and justified*	No single failure shall have critical or catastrophic consequences (i.e. loss of mission). Redundancy or compensation concepts shall be considered to minimise consequences of single point failures. *C: any deviation with respect to this requirement shall be identified and justified for approval. Failure tolerance does not need to be applied to: primary structures, load-carrying structures, structural fasteners, load-carrying elements of mechanisms, and pressure vessels. In these cases, the requirements of design for minimum risk shall be applied.*

Table 23-1: Change Request to MRD CO-30 Requirement

23.2 Risk Management Process

Risk management is an organised, systematic decision making process that efficiently identifies, analyses, plans, tracks, controls, communicates, and documents risk in order to increase the likelihood of achieving the project goals. The procedure comprises four fundamental steps RD[36]:

- Step 1: Definition of the risk management policy which includes the project success criteria, the severity & likelihood categorisations, and the actions to be taken on risks
- Step 2: Identification and assessment of risks in terms of likelihood and severity
- Step 3: Decision and action (risk acceptance or implementation of mitigating actions)
- Step 4: Communication and documentation.

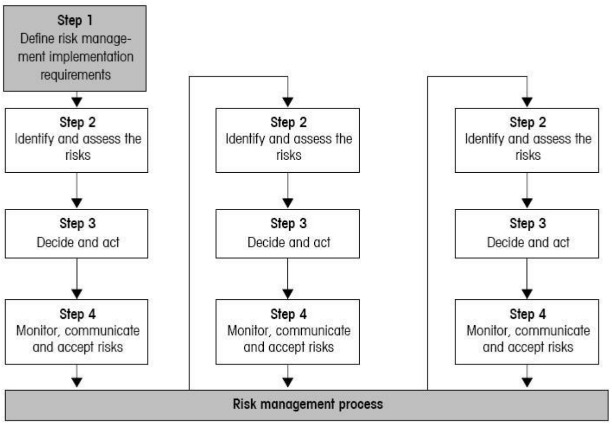


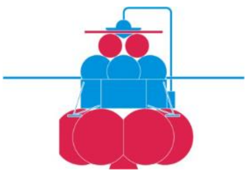
Figure 23-1: ECSS-M-ST-80C, 2008 Risk Management Process

23.3 Phobos Sample Return Risk Management Policy

The CDF risk management policy for Phobos Sample Return aims at handling risks which may cause serious science, technical, schedule and/or cost impact on the project.

23.3.1 Success Criteria

The success criteria with respect to the science, technical, schedule, and cost objectives are presented in Table 23-2:



Domain	Success Criteria
Science	• The ESA elements contribute to the understanding of the formation of the Martian moons Phobos and Deimos and put constraints on the evolution of the solar system ○ The ESA elements contribute to the return to Earth of >100g of Phobos regolith for scientific research. The ESA elements contribute to ensuring the preservation of the sample from its acquisition on the surface of Phobos, until its delivery for analysis to the receiving facility on Earth. ○ The ESA elements contribute to the characterisation of Phobos from a scientific point of view at global and local scale, and Deimos at global scale.
Technical	• The ESA elements contribute to the demonstration or maturing of technologies required for Mars Sample Return. • European cooperation elements* perform their respective functions successfully without failure.
Schedule	• The ESA elements meet the programmatic deadlines for launch in 2024 (2026 backup). • Considered technologies reach TRL 5 in 2016.
Cost	Cost at completion for the development of the ESA elements does not exceed the estimated budget (TBD M€).

Table 23-2: Success Criteria

It is to be noted that the European cooperation elements comprise the following: Earth re-entry capsule, lander, sample acquisition/transfer/containment system, science instruments, operations (except launch, ERV), and sample receiving facility.

23.3.2 Severity and Likelihood Categorisations

The risk scenarios are classified according to their domains of impact. The consequential severity level of the risks scenarios is defined according to the worst case potential effect with respect to science objectives, technical performance objectives, schedule objectives and/or cost objectives.

In addition, identified risks that may jeopardize and/or compromise the European contribution to the Phobos Sample Return mission will be ranked in terms of likelihood of occurrence and severity of consequence.

The scoring scheme with respect to the severity of consequence on a scale of 1 to 5 is established in Table 23-3, and the likelihood of occurrence is normalised on a scale of A to E in Table 23-4.

Severity	Science	Technical	Schedule	Cost
Catastrophic	Failure leading to the impossibility of fulfilling the mission's Scientific objectives	Safety: Loss of life, life-threatening or permanently disabling injury or occupational illness; Severe detrimental environmental effects. Loss of system, launcher or launch facilities	Delay results in project cancellation	Cost increase result in project cancellation
Critical	Failure results in a major reduction (70-90%) of mission's Science return	Safety: Major damage to flight systems, major damage to ground facilities; Major damage to public or private property; Temporarily disabling but not life- threatening injury, or temporary occupational illness; Major detrimental environmental effects Dependability: Loss of mission	Critical launch delay (TBD months)	Critical increase in estimated cost. TBD M€
Major	Failure results in an important reduction (30-70%) of the mission's Science return	Safety: Minor injury, minor disability, minor occupational illness. Minor system or environmental damage Dependability: Major degradation of the system	Major launch delay (TBD months)	Major increase in estimated cost. TBD M€
Significant	Failure results in a substantial reduction (<30%) of the mission's Science return	Dependability: Minor degradation of system (e.g.: system is still able to control the consequences) Safety: Impact less than minor	Significant launch delay (TBD months)	Significant increase in estimated cost. TBD K€
Minimum	No/ minimal consequences	No/ minimal consequences	No/ minimal consequences	No/ minimal consequences

Table 23-3: Severity Categorisation

Score	Likelihood	Definition
E	Maximum	Certain to occur, will occur once or more times per project.
D	High	Will occur **frequently**, about 1 in 10 projects
C	Medium	Will occur **sometimes**, about 1 in 100 projects
B	Low	Will occur **seldom**, about 1 in 1000 projects
A	Minimum	Will **almost never** occur, 1 in 10000 projects

Table 23-4: Likelihood Categorisation

23.3.3 Risk Index & Acceptance Policy

The risk index is the combination of the likelihood of occurrence and the severity of consequences for a given risk item. Risk ratings of low risk (green), medium risk (yellow), and high risk (red) were assigned based on the criteria of the risk index scheme (see Figure 23-2). The level of criticality for a risk item is denoted by the analysis of the risk index. By policy high and medium risks are not acceptable and must be reduced.

Severity					
5	5A	5B	5C	5D	5E
4	4A	4B	4C	4D	4E
3	3A	3B	3C	3D	3E
2	2A	2B	2C	2D	2E
1	1A	1B	1C	1D	1E
	A	B	C	D	E
					Likelihood

Figure 23-2: Risk Index

23.4 Risk Drivers

A number of risk drivers have been considered in the identification of specific risk items. These are gathered below:

- New technology
- Environmental conditions (radiation, micrometeoroid, dust, vacuum, extreme temperature gradients, etc.)
- Design challenges
- Reliability issues, single point failures (SPFs)
- Centralized vs. decentralized avionic architecture and its implications on reliability, failure tolerance, complexity of interfaces, individual module complexity and development risk
- Major mission events (launch and Earth escape, Mars capture, Phobos landing, Earth re-entry, etc.)
- External (uncontrolled) risks inherent to international cooperation missions (e.g. interfaces)
- ITAR Restrictions.

23.5 Top Risk Log

Top risk items have been identified based on their impact on technical/science, schedule, cost, and safety. Please refer to Table 23-6 for a complete list of identified top risks and their corresponding suggested mitigating actions. Risk index results are summarised in the Top Risk Index Chart below:

Severity					
5			COS_02	COS_01	PRO_05
4		TEC_04/05	TEC02/03/07/15	TEC_01/08/09/11/12/13/16 PRO_01/02/03/04	TEC_10
3				TEC_06/17/18	
2					TEC_14
1					
	A	B	C	D	E
					Likelihood

Table 23-5: Top Risk Index Chart

Risk ID	Mission Element	Risk index	Risk scenario	Cause	Mitigating Action 1	Mitigating Action 2	Mitigating Action 3
Technical/Science							
TEC_01	ERC	4D	Challenging ERC high speed impact landing. Survivability of sample canister.	• No parachutes baselined for ERC re-entry and descent. • Possible off-nominal impact axis. • Weakness in ERC closure/locking. • Low TRL of energy absorbing materials (TRL 3).	Design of crushable structure with margins to ensure survivability of sample canister (principal and off-nominal impact axes).	Assess static/dynamic strength of crushing material at elevated temperatures.	• Consider use of parachutes to slow down capsule taking into account cost and technology development issues. • Prepare development plan for crushable materials. Invest in technology development and testing. Consider margins in schedule.
TEC_02	ERC	4D	ERC is not found/detected after landing.	• Large landing dispersions • Landing outside predicted area.	Incorporate beacons in ERC design.	Study alternative on-ground localization methods (e.g. radar, seismology, etc.).	
TEC_03	ERC	4D	Challenging ERC stability in the transonic range.	Uncertainty in stability parameters in transonic range.	• Stringent control of CoM. • Spherical backshell.	Technology development plan to include aerodynamic test campaign.	Addition of supersonic parachute (low TRL).
TEC_04	ERC	4D	Critical separation(s) (lander-ERV, ERV-ERC) and entry sequence.	Single Point Failure(s). Collision risk after separation.	Single actuation and short duration events. All pyrotechnic devices should be equipped with redundant ESA standard actuators.	Specify sufficient margins in Lander/ERV, ERV/ERC ejection system to avoid risk of collision.	
TEC_05	ERC	4C	Damage to ERC TPS preventing a safe Earth re-entry.	Micrometeoroid penetration.	Investigate micrometeoroid environment, compute probability of impact and perform a damage assessment.	In case of high risk of micrometeoroid penetration consider shielding of ERC front shield if feasible from mass point of view.	Accommodation of ERC to minimise probability of micrometeoroid impacts on TPS.
TEC_06	ERC	4D	Increase in ERC heat shield mass.	• ERC Entry velocity at TPS material limit. • Turbulent heat fluxes not considered. • Uncertainties in convective and radiative heat fluxes. • Uncertainties in statistical material properties for TPS materials	Thermal-structural design and analysis based upon FEM will be insufficient – combined environment testing, with thermal gradients and mechanical loads is needed.	Experience/time required to develop a credible and validated series of FEM models for an integrated heat shield to assess various load cases.	Invest time in establishing an acceptable thermal-structural margins policy.

Risk ID	Mission Element	Risk index	Risk scenario	Cause	Mitigating Action 1	Mitigating Action 2	Mitigating Action 3
				(i.e. obtaining mechanical properties)			
TEC_17	ERC	3D	Challenging thermal-structural analysis for ablative materials.	Statistical material properties do not exist for most TPS materials. Obtaining mechanical properties (highly non-linear) across a wide temperature range is challenging and for TPS materials often produce large variations. Failure modes are poorly understood.	Thermal-structural design and analysis based upon FEM is insufficient – combined environment testing, with thermal gradients and mechanical loads is needed.	Experience/time required to develop a credible and validated series of FEM models for an integrated heat shield to assess various load cases.	Invest time in establishing an acceptable thermal-structural margins policy.
TEC_18	ERC	3D	Re-entry safety requirements impact on ERC design and mission concept including release sequence. Uncertainties in entry corridor restrictions for considered landing site. Impact on design changes/schedule.	• No re-entry safety requirements in MRD. • Unknown entry corridor restrictions for considered landing sites.	Early Identification of re-entry safety authority and related re-entry safety requirements.	Investigate entry corridor restrictions for considered landing sites ahead of the start of Phase-A.	Perform detailed re-entry risk assessment.
TEC_08	Lander	4D	Challenging close proximity, landing, and surface operations resulting in loss of mission. Landing site morphology (slope / rocks) exceeds the design requirements of the landing system.	• Phobos ephemeris and gravity potential not well known. • Large S/C size incl. solar arrays. • Presence of hazardous terrain conditions such as boulders, slopes or poor solar illumination conditions (bottom of steep crater). Surface hazards may not be detectable from previous observations. • Unknown soil properties. Soil resistance lower than expected. • Mechanical response of solar array at landing.	• Absolute navigation system for descent and landing phase to increase landing accuracy. Knowledge of gravitational field, object shape, surface topography and general composition ahead of close proximity operations. • Include a robust collision avoidance strategy.	Characterise surface and near-space environment ahead of close proximity operations to reduce risk. Final landing site selection during Phobos characterisation phase ahead of landing.	Further investigate mechanical response of solar array at landing. Fixed landing leg baseline.

Risk ID	Mission Element	Risk index	Risk scenario	Cause	Mitigating Action 1	Mitigating Action 2	Mitigating Action 3
TEC_09	Lander	4D	Mechanisms criticality (antenna pointing, landing systems, SA HDRMs, sample acquisition, transfer, and containment/sealing system (SATCs)).	• Single point failure. • Direct sun exposure (some parts not covered by MLI). • Uncertainties in force/torques during sampling depending on soil properties. • Subject to dust contamination. • Low technology readiness level.	• Risk reduction by increasing failure tolerance. • Implementation of independent mechanisms for independent functions. • Implementation of internal redundancies. • Minimise the number of robotic arm actuations (single sample).	• Investigate criticality of direct sun illumination for mechanisms and decide whether mechanisms need to be thermally protected. • Assess criticality of dust environment and its impact on mechanisms operation (e.g. robotic arm) and ERC sealing.	• Invest in technology and testing. • Appropriate testing/qualification strategy with wide range of soil properties. • Investigate alternative back up sampling tools.
TEC_11	Lander	4D	Exceeding Lander mass budget/ volume constraints impact on mission feasibility.	Stringent launch mass constraints. Limited clearance with Proton fairing (fixed landing leg system).	Insert sufficient equipment and system mass margins according to maturity levels as per CDF policy.	Consider solid mission alternatives with little or no science at Deimos	
TEC_12	Lander	4D	Failure to deliver required ΔV during critical mission phases.	• Reaction control system failure, critical SPFs. • Controllability issues.	Baseline configuration providing greatest controllability and simplicity in design.	Baseline equipment with successful heritage (high TRL). Baseline critical single point failure free design (upstream from engine(s))	Mission options with shorter lifetimes and lower number of engine cycles preferred.
TEC_13	Lander	4D	Failure, temporary loss, or performance uncertainties of GNC systems.	• Blinding, high star richness. • Equipment failure due to high radiation environment. • GNC WAC insufficient image quality for navigation/ground feature tracking. • Insufficient altimeter accuracy/range. • Tracking algorithm may not be robust to Phobos shape and rapidly varying illumination.	Robust redundant design. Single point failure free.	Select equipment with flight heritage (high TRL).	• Early environmental qualification. • Extensive simulations campaign.
TEC_14	Lander	2E	Dust contamination impact on optical/mechanical equipment performance	Phobos environment, specifically during and post-sampling operations.	Assess criticality of dust environment and its impact on optical/mechanical equipment performance.	Baseline free-fall landing from an altitude that is sufficient to minimise dust impact on optical mechanical equipment.	
TEC_15	Lander	4C	Sample is contaminated during landing and	GNC thruster exhaust plume impingement on	Design free-fall and thruster configuration in	Monopropellant (hydrazine) lander propulsion system	

Risk ID	Mission Element	Risk index	Risk scenario	Cause	Mitigating Action 1	Mitigating Action 2	Mitigating Action 3
			sampling operations.	sampled Phobos soil.	accordance with contamination requirements.	preferred to bipropellant to minimise contamination impact.	
TEC_16	Lander	4D	Complex sample acquisition system design and qualification.	• Uncertainties in Phobos soil properties (i.e. density, temperature, compression strength...). • Test in vacuum environment.	Further investigate Phobos soil properties with science team to better define ranges of mechanical properties and gain confidence in the robustness of the qualification strategy.	Sampling mechanism design able to cope with range of soil properties.	• Multiple sampling attempts. • Phobos soil characterisation processing on ground ahead of descent trajectory planning.
TEC_10	European Cooperation Elements	4E	Uncertainties in mass budget and interface requirements.	Limited data available on mass or interface requirements.	Clarify mass and interface requirements for Lander-ERC in advance of the beginning of Phase A.	Establish a good working relationship with partner agency.	
TEC_07	Ground	4C	Time delay for communication and ground processing leads to loss of mission during critical Phobos operations (descent and landing, safe mode).	Challenging Earth-S/C distance during Mars arrival and Phobos ops.	Early consolidation of realistic operation scenario for Phobos approach and D&L phases.	• Identify time critical constraints. • Define autonomy strategy during approach, descent, and landing including S/C safing during off-nominal scenarios (i.e. S/S failure, unexpected attitudes after surface contact, etc.).	Alternative trajectories which reduce Earth-S/C distance during Mars/Phobos critical ops.
Schedule							
PRO_01	ERC	4D	Delays in schedule and performance uncertainties of ERC critical technologies: • Low/mid density European ablative materials for ERC TPS • Energy absorbing materials • Beacon	• Ablative Materials: o Low TRL o Development challenges. o Ablative materials manufacturing complexity o Limited capability of ground facilities (arc jet) for ablative material testing o Low number of available testing facilities. Even an ideal ground test facility will not fully replicate flight environments forcing difficult	• Ablative Materials: Closely monitor European low density ablative carbon phenolic material development: ESA TRP "DEAM 2" Development of European Ablative Material. Additional investment in the TDA to achieve technology readiness objectives.	• Ablative Materials: Restarting the manufacturing of previous TPS materials takes significant time and resources. Significant fabrication experience is required to produce quality and consistency >Investment required to establish necessary infrastructure. Selection of experienced TPS manufacturer.	All ERC critical technologies: • Prepare realistic development and testing plan. • Plan schedule accordingly. • Insert margins in schedule. • Drop tests for TPS and crushable materials. .

Risk ID	Mission Element	Risk index	Risk scenario	Cause	Mitigating Action 1	Mitigating Action 2	Mitigating Action 3
				ground-to-flight traceability efforts. Prone to high down time. • Energy Absorbing materials are currently at TRL 3 • Beacon is a new development at TRL 2.			
PRO_02	Lander	4D	Low TRL of proximity GNC, landing system, sample acquisition, transfer & containment system (SATCS), and payload equipment impact on project schedule and technical performance	Low TRL of: • Wide angle camera (TRL-4) • Altimeter sensor (TRL-3) • Image processing/GNC algorithms (TRL 4) • SATCS (TRL 2-4) • landing legs/damper and deployment mechanisms (TRL 2-4) • VIS/NIR sensor in the 0.4-3.3nm range.	Some technology developments are ongoing..	Additional investment in technology and testing required to meet TRL objectives.	
PRO_03	European Cooperation Elements	4D	Unavailability of existing test facilities for critical lander and ERC technologies.	• Limited availability of facilities and expertise. • Facilities are prone to high downtime.	Plan schedule accordingly. Consider margins in schedule.	Select backup facilities. Increase investment in European technology development efforts.	
PRO_04	European Cooperation Elements	4D	External risks (delays, technical challenges) impact on the development cost and schedule of European cooperation elements.	International cooperation mission with multiple external risks which are uncontrollable for ESA.	Minimise number and complexity of module interfaces.	Establish a close cooperation with partner agency with regular progress meetings. Create a trusting and open environment enabling improved communication flow and quicker problem notification.	Adequate funding of dedicated ESA interface team with partner agency (preferably fluent in Russian).
PRO_05	European Cooperation Elements	5E	Delays in schedule and technical showstoppers as a result of ITAR export regulation restrictions.	The Directorate of Defense Trade Controls (DDTC) of the U.S. State Dept. approves the export licenses required to launch U.S. ITAR components on Russian launch vehicles. At the time of writing this approval has	Early identification of ITAR components to evaluate the possible extent of the impact and prepare contingency plan.	U.S. administration is in the process of relaxing export controls for U.S. communications satellites and related components. The final rule on a proposed list of space-related technologies to be removed from the U.S. Munitions List	Insert margins in schedule.

Risk ID	Mission Element	Risk index	Risk scenario	Cause	Mitigating Action 1	Mitigating Action 2	Mitigating Action 3
				been suspended.		is expected by the end of 2014.	
Cost							
COS_01	European Cooperation Elements	5D	Cost increase results in project cancellation.	International cooperation mission with multiple external risks which are uncontrollable for ESA.	Limit design to minimum required to complete mission.	• Maximize re-use of available technologies. • Discard technology solutions which do not comply with TRL requirements and are thus not compatible in terms of schedule and cost.	No commitments on additional science payloads.
COS_02	European Cooperation Elements	5C	Mission is rated CAT-V restricted Earth return impact on cost and schedule.	• Planetary protection category not yet confirmed for Mars' Moons. • Impact on design, AIVT procedures, and documentation (including sample receiving facility).	Determine/confirm as soon as possible the planetary protection category as category V unrestricted Earth return.		
Safety							
SAF_01	Lander	5B	Safety risk to ground crew during hydrazine filling operations and spacecraft handling. Fire, explosion (high pressure), contamination.	Uncontained hydrazine leak.	Specific competences and experience to comply with the applicable safety regulations (national, launcher, and launch site).	European major prime contractors have the required experience and know-how on hydrazine handling.	

Table 23-6: Top Risk Log

23.5.1 Risk Log General Conclusions

- High risks are typical of a phase A project. Areas with lack of definition or little previous experience pose a priori more risk to the mission and therefore are the ones with more risk reduction potential
- Experience shows that all risk items with a critical risk index (red/yellow area) must be analyzed and proposals for risk treatment actions elaborated
- In the end, ideally all risk items should reach a level of justifiable acceptance
- The risk management process should be further developed during the project definition phase in order to refine the risk identification/analysis and provide evidence that all the risks have been effectively controlled.

23.6 Staging Options Comparative Risk Assessment

A (qualitative) comparative risk assessment was performed for staging options 1 through 4 following a phased mission approach. Technical risk was estimated for each staging option during each mission phase. Staging options were assessed locally by mission phase and then globally by adding all risk contributions to the different mission phases. A score of one (1) was given to low risk, a score of two (2) to medium risk, and a score of five (5) to high risk per option and mission phase. Weights were assigned to each mission phase based on expert judgement and depending on its complexity and contribution to the overall mission scenario. A weight factor of 2 (w=2) was assigned to the Earth escape, MOI, Phobos D&L, Phobos ascent/Mars escape, and Earth re-entry phases. All other mission phases were assigned a weight factor of 1 (w=1). On the other hand, development risk was assessed per staging option.

23.6.1 Staging Options Definition

Below is a summary table of the staging options that were considered in the comparative risk assessment. The table shows which module performs the (high) thrust engine manoeuvres in a given phase:

	High Trust Engine Maneuvers								
	Earth-Mars Cruise Correc.	MOI	TOA-1 Deimos	TOA-2 Deimos	Deimos-Phobos	Around Phobos	Phobos D&L	Phobos Ascent & Mars Escape	Mars-Earth Transfer Correc.
Option 1	LM	LM	LM	LM	LM	LM	LM	ERV	ERV
Option 3	PM	PM	PM	PM	PM	ERV	ERV	ERV	ERV
Option 4A	PM	PM	LM	LM	LM	LM	LM	ERV	ERV
Option 4B	PM	PM	PM	LM	LM	LM	LM	ERV	ERV
Option 4C	PM	PM	PM	PM	LM	LM	LM	ERV	ERV
Option 2/4D	PM	PM	PM	PM	PM	LM	LM	ERV	ERV

Table 23-7: System Options Considered in the Comparative Risk Assessment

23.6.2 Analysis, Results, and Conclusions

Results show that a higher number of engine cycles implies a higher probability of failure. Therefore, staging options such as 4b and 4c with a balanced spread of propulsion functionalities across modules are lower risk. On the other hand, integrating the propulsion module functions in the Lander module (option 1) would lead to a more complex and higher mass/volume Lander. This would have implications on landing risk and may impact the complexity of the sample acquisition and transfer system.

The development risk of the European elements was assessed to be high for option 1, moderate for options 2 and 4 and, low-to-moderate for option 3. The assessment was based on functional complexity, heritage in Europe, and technology readiness level.

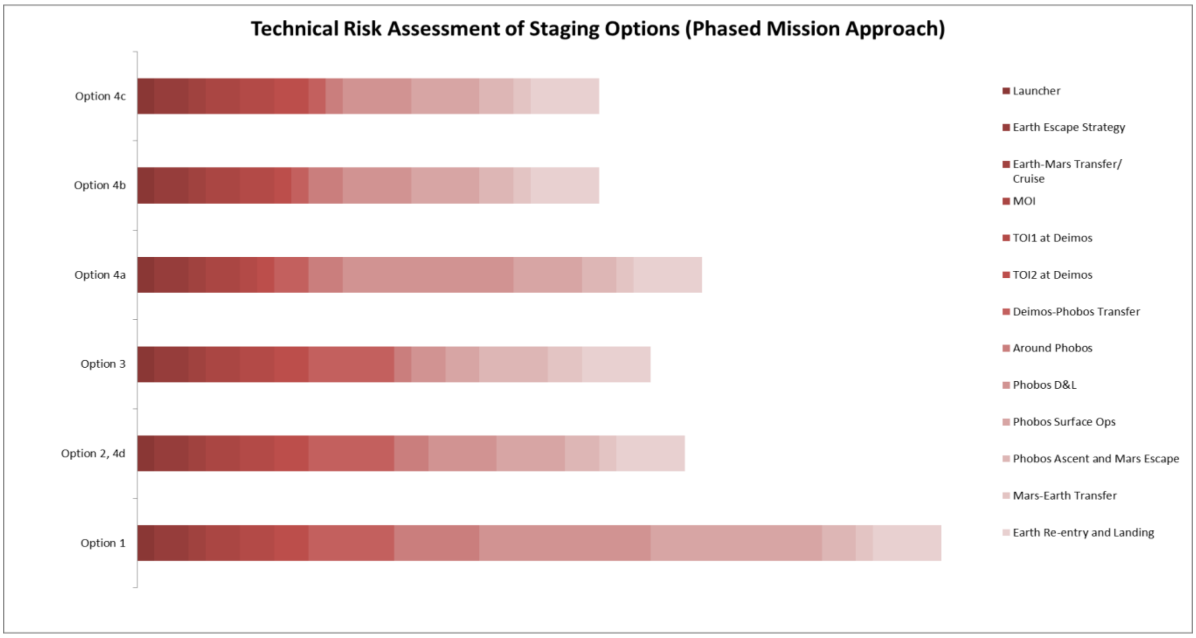


Figure 23-3: Results of the Staging Options Comparative Risk Assessment

24 ESA SAMPLING CHAIN

24.1 Requirements and Design Drivers

24.1.1 Design Drivers

Bringing a sample back to Earth from Phobos can only be established by protecting the sample in a light Earth Re-entry Capsule (ERC). A system with the ability of taking a sample of regolith and bring it into the ERC is required in this mission. This system will consist of a robotic arm and a sampling tool. There are different ways to use a robotic arm and a sampling tool. These different ways can be described in different sampling chains. A trade-off was made to find a chain that fits the requirements the most, from ESA perspective only.

24.1.2 Requirements

The following table summarises the different mission and subsystem requirements which are relevant for the sampling chain.

SubSystem requirements		
Req. ID	**STATEMENT**	**Parent ID**
MI-10	The mission shall return approximately 100g of loose material from the surface of Phobos	
MI-220	The mission shall perform 3 types of surface operations in order to fulfil requirements in the Science Requirement Document: - Sampling point selection and characterisation - Sample acquisition and transfer to ERC - Post-sampling science measurements	
MI-260	Once landed, the mission shall allow the Ground to select the sampling location within the sampling tool range	
MI-270	The mission shall provide the possibility to the Ground to check that the collected sample is suitable before transfer to the ERC	
MI-280	The mission should implement on-board automatic procedures to perform contingency sampling and lift-off operations in case of communication failures with the ground	
CO-30	The Composite shall provide single point failure tolerance. Redundancy concepts shall be considered to minimise consequences of single point failures *C: any deviation with respect to this requirement shall be identified and justified*	
CO-60	In the Composite design, only technologies that can be assumed to be at TRL 5 at the start of the mission implementation phase shall be considered when defining the mission architecture	
LM-60	The LM shall allow the sampling, transfer to ERC and sealing of the sample	

24.2 Assumptions and Trade-Offs

24.2.1 Trade-Off: Sampling Method

The following, presents a trade-off between the different sampling concepts considered in the frame of the CDF study on Phobos Sample Return. This trade-off has been used to justify the choice of the concept adopted as "ESA sampling chain".

24.2.1.1 Sampling procedures

Different sampling procedures are illustrated in Figure 24-1 that represents a tree of possible options. The different scenarios are further explained later on.

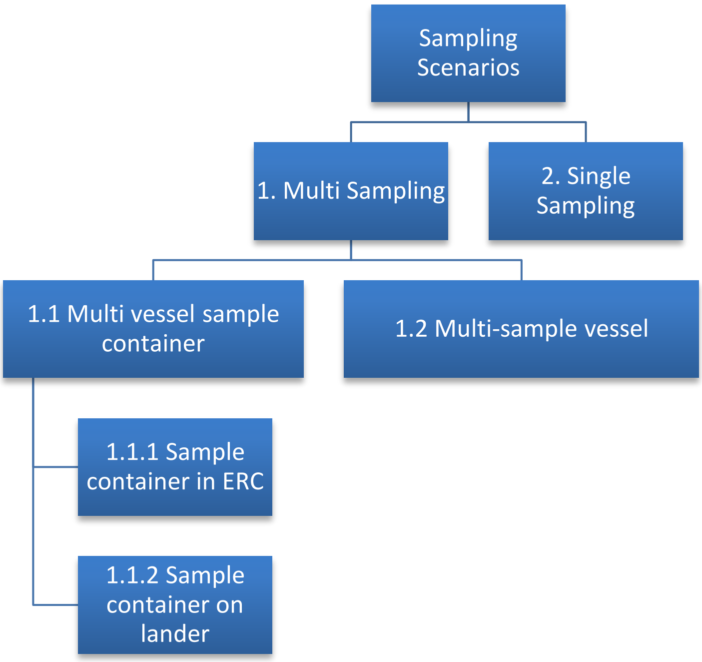


Figure 24-1: Sampling procedures

24.2.1.2 Sample container in ERC

The first multi-sample solution considered has a sample container permanently placed in the ERC, holding all the samples. The robotic arm has to open the ERC, take a sample from the Phobos soil and transfer it all the way to the ERC. The latter two actions have to be repeated for all the samples the arm has to take. Subsequently, the arm has to close the ERC. Figure 24-2 gives visualisation of the setup.

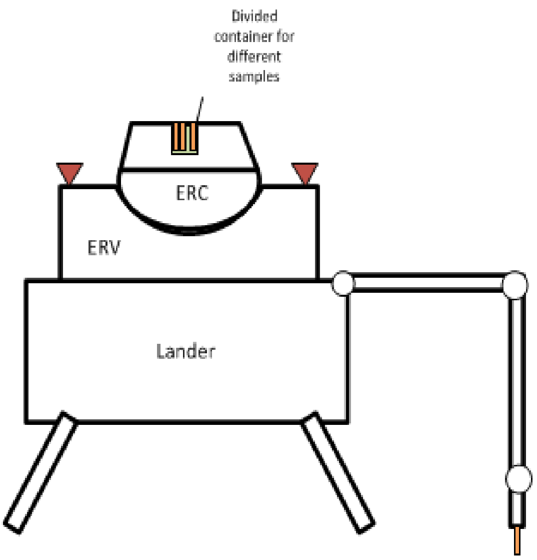


Figure 24-2: Schematic of the sample container in ERC

24.2.1.3 Sample container on the lander

A second multi-sample solution assumes a sample container placed on the lander, which is used to collect all the samples as close as possible to the surface. The advantage in this concept is that the arm does not have to travel the whole distance from the ground to the ERC for every sample. This method will decrease the time it takes to collect all the samples.

After the arm has collected all the samples, the arm transfers the sample container from the lander to the ERC and closes the ERC. An extra tool is necessary at the end of the robotic arm to grasp the sample container. The process is illustrated in Figure 24-3. On the left, the arm is collecting the samples and is transfers these samples to the container on the lander. On the right, the arm is transferring this sample container to the ERC.

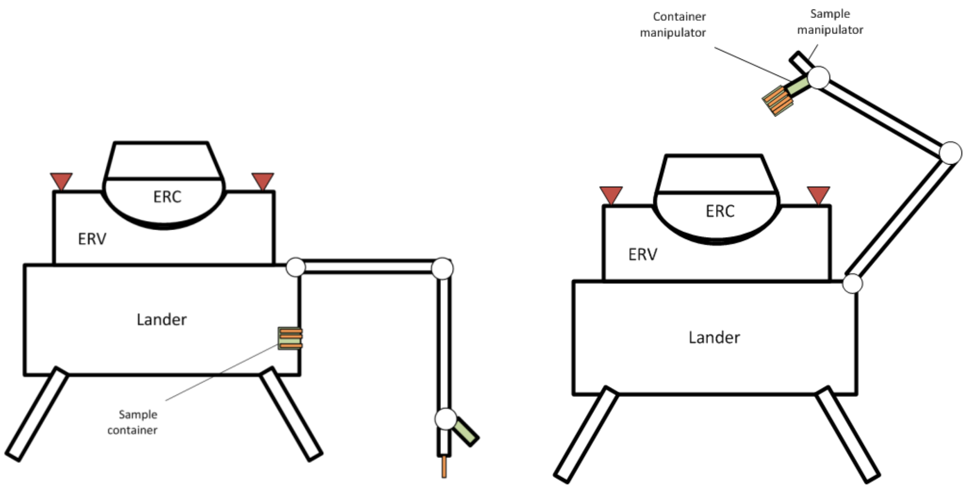


Figure 24-3: Sampling container on the lander

24.2.1.4 Multi-sample vessel

The difference with the previous sampling chains lies within the fact that the all the samples are collected in the same vessel. Thus there is no sample container. The arm takes different samples from different locations and those samples will end up mixed in the vessel.

24.2.1.5 Single sampling

In singles sampling the arm takes only one sample from the Phobos surface. This means that the arm reaches down once to collect a sample and transfers it up and inserts it into the ERC. The following illustration visualizes the single sampling.

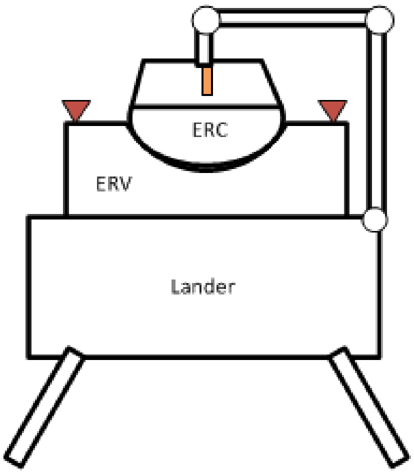


Figure 24-4: Single sampling

24.2.2 Trade-Off Criteria

The following criteria have been considered in the trade-off:

1. Technical Risk related to operations, additional functions and reaction forces
2. Time: duration of operations
3. Mass penalty of each solution related to
    a. extra structure on the return capsule
    b. extra fuel needed for stable operation
4. Energy required by arm operations & reactions force compensation

The trade-off was purely technical and quality of science was not included, although there are some concepts whose complexity appears not to be justified by the added quality of science (from ESA point of view).

24.2.3 Assumptions

In the trade-off, some assumptions were taken into account. These assumptions are summarised below.

- **Hold down thrusters:** In the negligible gravity of Phobos (0.0084 - 0.0019 m/s^2), every time the robot arm takes a sample, the lander takes off due to reaction forces that are bigger than the gravity pull. Figure 24-5 shows the development of contact forces involved in sampling without hold down thrusters, with a sample force of 3N on the Phobos soil.

- The arm is able to maintain contact by compensation for 10cm and 3 seconds. After three seconds the arm will loose contact with Phobos because the lander takes off due to the reaction forces. Therefore thrusters are used to keep the vehicle in place. The force of the thrusters compresses the structure of the lander platform.
- When shutting down the thruster, the lander will likely take-off anyway, due to the elasticity in the system (this needs further investigation). This risk should be taken into account while choosing the sampling procedure.

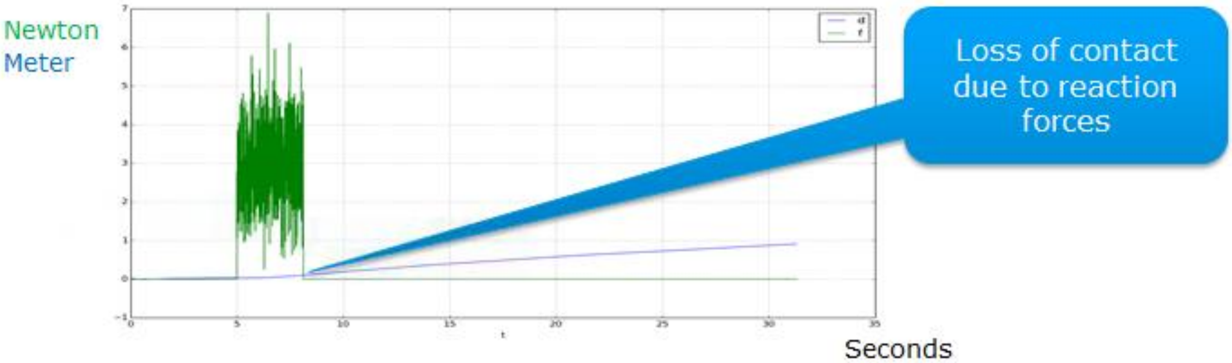


Figure 24-5: Simulation without reaction forces

- **Operations (risk)**: The number of individual operations in each sampling method is directly considered as a measure of risk. All methods are made of a sequence of operations and a problem in any operation prevents the accomplishment of the method and is therefore a potential mission failure.
- **Time**: Transferring one sample from Phobos to the ERC corresponds to one unit of time. Transferring a sample will be assumed as 0.5 units in time in case the sample container is mounted on the lander. Closing or opening operations corresponds to 0.2 units in time.
- **Mass of structure**: The mass penalty of the structure related to the different sampling methods has been estimated in relative terms. In order to keep samples separated from each other, extra structure is required, which implies extra mass. The more samples are required, the more tubes will have to travel back to Earth, and the more mass has to travel back to Earth with the ERC. In the following a calculation of the mass of tubes, related to the cross section of the sample, is shown. The calculation estimates the relative penalty of each sample method.
- **Mass of Fuel**: Taking a sample requires energy/fuel (and hence mass) to compensate the reaction forces. The amount of force and fuel needed can be estimated in relative terms as being proportional to the cross section of the sample.

24.2.4 Calculations Related to Cross Section of the Sample

Both, fuel consumption for reaction force compensation and mass of structure to keep the samples separated are related to the surface in the cross section of the sampling tube. Multisampling automatically means more surface area and consequently more fuel consumption and mass. In order to be able to compare different sampling methods, calculations are made to find a scaling factor between multisampling and single sampling. Multisampling is assumed to be 10 samples.

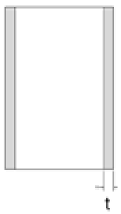


Figure 24-6: Dimensions vessel

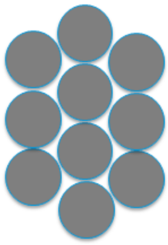


Figure 24-7: Multi sampling area

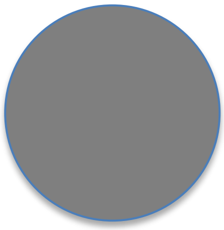


Figure 24-8: Single sampling area

- r =radius small samples
- R= radius one big sample
- T= thickness of sample tool.
- n= 10 samples

Assumption= As = Am

$$As = Am$$

$$\pi * R^2 = n * \pi * r^2$$

$$r = \frac{R}{\sqrt{n}}$$

Assumption: Force is linear related with outline $F \approx outline$

$$Am = 2 * \pi * r * n * t$$

$$As = 2 * \pi * R * t$$

$$\frac{Am}{As} = 3.16 \Rightarrow Am = 3.16 * As$$

Figure 24-9: Numerical trade-off

Energy multisampling = 3.16 * Energy single sampling

Mass multisampling = 3.16 * Mass single sampling

24.2.5 Operation and Trade-Off Criteria

The trade-off involving the different sampling types is documented in the following table. In order to shorten the table, operations which are repeated are replaced with three dots.

	1.1.1 Sample container in ERC	1.1.2 Sample container on Lander	1.2 Multi sample vessel	2 Singe sampling
Operations	1) Opening ERC 2) Sampling 3) Transfer sample to ERC 4) Close sample vessel 5) Sampling 6) Transfer sample to ERC 7) Close sample vessel … 32) Close ERC	1)Opening sampling Container 2) Sampling 3) Transfer sample to container 4) Close sampling vessel 5) Sampling 6) Transfer sample to container 7) Close sampling vessel … 32) Close Vessel (Gastight) 33) opening ERC 34) transferring container to ERC (extra manipulator+ oversizing) 35) Close ERC	1) Opening ERC 2) Sampling 3) Transfer sample to ERC 2) Sampling 3) Transfer sample to ERC … 22) Close ERC	1) Opening ERC 2) Sampling 3) Transfer sample to ERC 4) Close ERC
Operations (risk)	**32**	**35**	**22**	**4**
Total Fuel	**3.16**	**3.16**	**3.16**	**1**
Total Time	**12.4**	**8.4**	**10.4**	**1.4**
Total Mass	**3.16**	**3.16**	**1**	**1**

Table 24-1: Disadvantages for each case

It is apparent that single sampling is a clear winner of the trade-off, as it has lower values for every single criterion. It is reminded that the science value criterion was not included in this trade-off, since all concepts allow to fulfil the science requirements considered for this study.

24.2.5.1 Disadvantages for each case

The following tables compare non-numerical advantages to disadvantages.

1.1.1 Sample container in ERC	**1.1.2) Sample container on Lander**	**1.2) Multi sample vessel**	**2) Singe sampling**
Disadvantages	**Disadvantages**	**Disadvantages**	**Disadvantages**
Several extra manoeuvres: The arm needs to cover a **long distance** between the ground and the vessel for **several times**. This takes some **time** and more **chance for failures**.	Several extra manoeuvres: This takes some **time** and more **chance for failures**.	Several extra manoeuvres: This takes some **time** and more **chance for failures**.	We have only one sample
Separate samples means separate containers => more structural **mass** to bring back to earth	Separate samples means separate containers => more structural **mass** to bring back to earth	No different layers in the sample. (no depth study)	
Every sampling operation requires the lander to re-land -> extra fuel -> **+complexity** and **+mass**	Every sampling operation requires the lander to re-land -> extra fuel -> **+complexity** and **+mass**	Having the risk of taking off and landing again due to reaction forces	
Having the risk of taking off and landing again due to reaction forces	Need for an extra or multifunctional gripper	Quality of science if all the samples are mixed? Example: Desert	
	Requirements for the arm are more heavy		
	Extra manoeuvres = more chance for failure		
	Having the risk of taking off and landing again du to reaction forces		

Table 24-2: Trade-off disadvantages

1.1.1 Sample container in ERC	1.1.2) Sample container on Lander	1.2) Multi sample vessel	2) Singe sampling
Advantages	**Advantages**	**Advantages**	**Advantages**
Separate samples do not mix.	Separate samples do not mix.	Less mass to transport to earth (no separation mass)	Only four simple operations
There is no need to transfer the container vessel. => no need for an extra manipulator/tool	sampling becomes a easier task due to smaller movements from the ground to the vessel		Only one time the risk of taking off and landing again due to reaction forces
			Less mass to transport to earth
			Layered sample study could be possible.
			There is less fuel needed to counter the reaction forces.
			Minimum operations = minimum risk

Table 24-3: Trade-of advantages

24.2.6 Conclusions

Multisampling makes the mission more risky (more operations) and more difficult (extra mass and fuel).

Therefore ESA proposes for this study a robotic arm which is based on the single sampling concept, as was the case for the Phootprint studies.

24.3 Baseline Design

24.3.1 Robotic Arm

See the chapter about Robotics for the proposed Robotic Arm design.

24.3.2 Sampling Tool

The trade-off about the sampling method shows the preference of a single sampling concept. Therefore the sampling tool shall be capable of collecting and storing the full volume of sample particles. In general, to remove at-once from the ground a volume of about 100 g can require relatively high applied forces, since the soil can have a certain compactness. Devices that perform this task are typically drillers, corers, scoops etc.

The forces can be significantly reduced if only the first and relatively soft layer of sand is collected, and the needed volume reached by sliding the sampling tool on the ground.

Such a solution can be pursued if it is not needed to physically separate different samples.

In the following section, an example of conceptual design of the sampling tool is shown (from RD[20]). It basically consists of a set of 2 or 3 rotating bristles which remove and lift-up the regolith from the soil into the sample container.

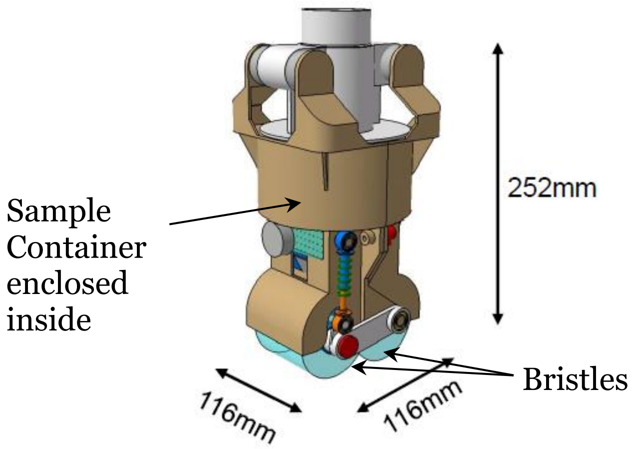


Figure 24-10: Example of a conceptual design of the Sampling Tool (Airbus DS)

A clear advantage of this solution is in the fact that the collection and transfer of the soil sand occurs within the same operation and is done by the same mechanical parts.

The bristles can be soft enough to adapt to the shape and roughness of the terrain without the need of a high pushing force from the robotic arm. The size of particle that can be collected can be up to several mm. The actual performance depends on the shape of the bristles, and it is under investigation.

An early prototype developed for ESA demonstrated operations with 10N contact force, a mass of 4.1 kg (excluding the sample container), and power consumption probably in the range of 10-20 W. The TRL can be assessed as between 3 and 4 at the moment.

Particular care must be taken in designing the mechanisms in order to prevent that particles of dust, lifted up by the bristles, can block parts in relative motions. Suitable sealing should be employed. Special attentions deserves the doors which will close the sample container once the sampling is concluded, and any jamming or uncompleted closure of mating faces prevented. Probably, the actual motion of the particles moved by the bristles in very low gravity has to be studied and risk-mitigation measures developed.

The mechanisms are also required to work in an environment at temperatures approximately between 100 and 300 K. The lower temperatures are not compatible with grease lubrication on bearings and gearboxes, therefore a need for heating the mechanisms to a range of -40 deg/+60 deg is needed, or the use of dry-lubrication is required. Also thermo-elastic effects and change of material properties must be carefully investigated.

The sampling tool must be provided with sensors to gauge the amount of samples collected and measure the force applied by the robotic arm on the soil. The monitoring of the temperature of the sample could also be needed for scientific reasons. Other sensors should provide information about the status of the mechanism, to understand if failures or jams have occurred.

After the sampling operation terminated, the two halves holding the brushes can open, to allow the robotic arm place and secure the sample container into the ERC.

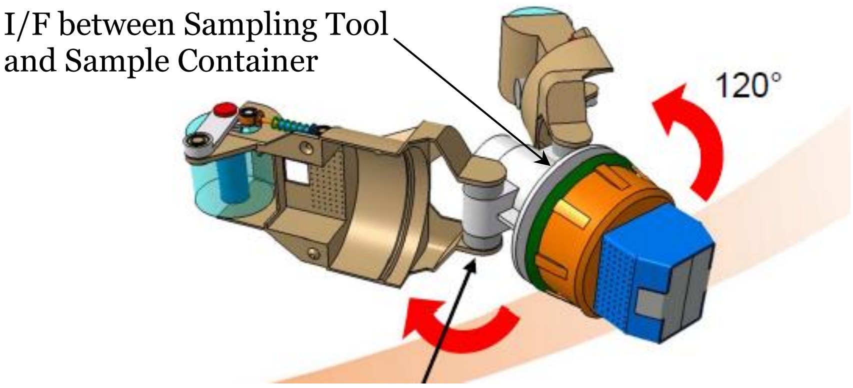


Figure 24-11: Conceptual design of the Sampling Tool (Airbus DS): opening of the halves to release the Sample Container (in blue)

See the also the description of the sample container in the Mechanisms chapter.

24.4 List of Equipment

See the Mechanisms chapter for a list and mass budget for the sampling tool and the robotic arm.

24.5 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Included in this table are:

- Technologies to be (further) developed
- Technologies available within European non-space sector(s)
- Technologies identified as coming from outside ESA member states.

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
Sampling Tool	Collection and storage of sand-like soil by means of rotating bristles.	Airbus DS / AVS, TRL 3/4	Yes	Several activities lead by ESA are developing breadboard-level models.
Robotic arm	Transferring a sample from a moon to the ERC.	Selex TRL4 by 2016	Yes	

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25 IKI SAMPLING CHAIN

25.1 Study Assumptions

IKI suggests to consider the implementation of double manipulator system. This approach is quite useful for multi-sampling concept. For science reasons the obtaining of a raw of samples has several advantages in comparison with one sampling attempt because provides to carry out investigations on different type of the moon soil thus leading to the better science return of the mission.

The 1st manipulator (MM) is supposed to deal directly with the Phobos surface for sampling. MM consists of Robotic Arm (RA), Sampling Device (SD) and Control Unit (CU).

This approach is inherited from Phobos Soil Mission (PhSM). On that spacecraft a 950 mm robotic arm was deployed and the sampling device collected 2.5 cm^3 of the soil. The mass of MM PhSM was 2.2 kg including SD. The SD weighted 200 g (at present this could be reduced to less than 200 g). Maximum power consumption was 15 W (when all motors of the MM were active, 5 W/drive). MM had 5 DoF (4 motors, SD – 1 DF). An accuracy of less than 1 mm was attained for the SD positioning. The PhSM MM had passed the complete test program and was mounted on the spacecraft.

The similar MM is under development for Lunar Mission (LuM). Its MM has the special implement for soil cleaning before sampling procedure like a scoop. By means of this scoop it is possible to extract soil from 150 mm below the surface. LuM MM has passed the complete test program at present time and is ready for mounting.

The 2nd manipulator or Transportation Tool (TT) for the Boomerang/Phootprint would contain a Robotic Arm, Sampling Container (where Phobos soil should be placed) and Control Unit. The main purpose of the TT is to transport and place sample container from MM outloading point into ERC. Therefore the TT needs to have only 1-2 DoF. Because of the low number of DoF the TT would have high accuracy at sample trans-shipment and at sample container outloading into ERC. Moreover this device could be considered as redundant mechanism for sampling in case of any incidents with MM. The program of emergency sampling could imply to grab any soil at the point the TT could reach. The redundancy option of TT is increased reliability of the sampling procedure.

According to the double manipulator concept MM is easy to manufacture. It would be lighter than its precursors (PhSM and LuM) and subsequently requiring less power. The length of its robotic arm could achieve 1000-1200 mm. The implementation of only the one robotic arm concept leads to its lengthening, mass growth and complication of procedures algorithms.

25.2 Design Description

The proposed sampling chain should contain 4 basic elements:

1. MM (RA with SD on its tip) (ROSCOSMOS responsibility)

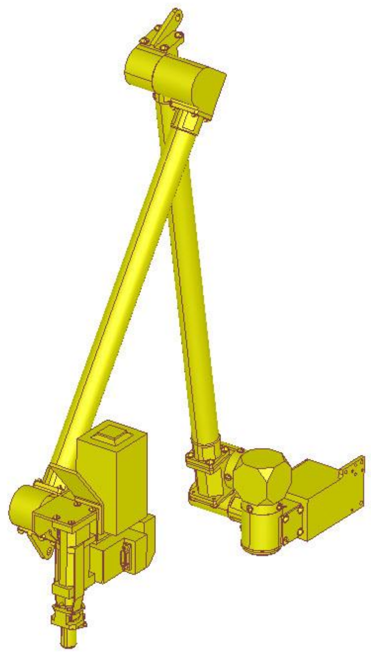


Figure 25-1: RA and SD of Phobos Soil mission (proposed for PhSM)

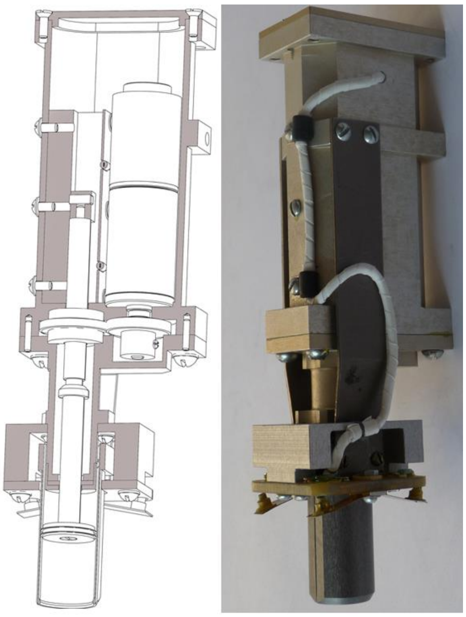


Figure 25-2: SD of Phobos Soil mission (proposed for PhSM)

2. Sample holder (ESA responsibility)

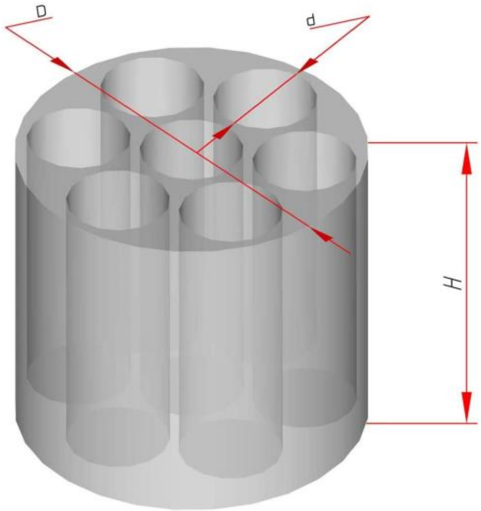


Figure 25-3: SC sketch

3. Instrument for sample holder closing/encapsulation (ESA responsibility)
4. TT (ESA responsibility).

The number of MM actions and their recording is defined and performed by video camera mounted at the tip of RA near to SD. After soil extraction, the sample should be trans-shipped from SD into the SC which may contain several cells for different samples. The preliminary calculation of SC and its cells is presented in Table 25-1.

	Sampling device			Sample container				No. of probes in one cell of container	Container loading time, min
	Inner dia. of sampling tubular, mm	Length dia. of sampling tubular, mm	Vol. of sampling tubular, cm^3	D, mm	d, mm	H, mm	Volume, cm^3		
Phobos Soil	13	22	2,5 - 3	50	14	120	100	5	180
Boomerang A	20	35	11	70	21	80	150	2	70
Boomerang B	25	45	20	85	26	50	140	1	35
Boomerang C	20	35	11	-	50	80	150	14	70

Table 25-1: Estimation of SC dimensions

The soil trans-shipment is accomplished in the point of mutually agreed position for MM and TT both. It is the position of their mechanical interfaces interconnection. For simplicity, soil trans-shipment reasons SC should be mounted at the tip of TT. Therefore IKI suggests ESA to take responsibility for the SC. It should be noted that SC would a priori have a more difficult mechanical interface with ERC. This circumstance could serve as additional factor to put SC and closing/encapsulation instrument under ESA responsibility for the simplicity of testing performance. IKI needs only coordinates of SC cells for testing procedures. Implementation of double manipulator complex allows both agencies (ESA and ROSCOSMOS) to participate in design, testing and operation of sampling chain.

25.3 Interface Requirements

The proposed ROSCOSMOS responsible parts need the following interface option:

1. Non-operational temperature: -70 +20 C
2. Operational temperature: -40 +20 C
3. Power supply +27V, 15 W (max)
4. Data interface: RS-485

Mechanical interface should be agreed at the following stages of the mission design. Preliminary estimations have shown the preferable transportation, non-operational and stand-by MM position as vertical.

25.4 Technology Requirements

The following technologies are required or would be beneficial to this domain:

Included in this table are:

- Technologies to be (further) developed
- Technologies available within European non-space sector(s)
- Technologies identified as coming from outside ESA member states.

Equipment and Text Reference	Technology	Suppliers and TRL Level	Technology from Non-Space Sectors	Additional Information
Main Manipulator	Transferring a sample from a moon to the ERC.	TRL 7	Yes	
Sampling Device	Collection and storage of sand-like soil by means of collet mechanism.	TRL 7	Yes	

26 CONCLUSIONS

26.1 Reference Design Conclusions

The study has concluded on a reference mission scenario with the following main characteristics answering the Reference Mission Architecture presented in Figure 26-1 as well as the baseline mission elements sharing presented in Figure 26-2.

- Russian Elements (PM & ERV) design based on Phobos-Grunt mission,
- European Elements (LM & ERC) design based on ESA Phootprint studies,
- Outbound double stage transfer with staging at Phobos arrival after Deimos visit,
- Joined European / Russian scientific payload in LM,
- 2 options for sample acquisition have been investigated (one allowing bulk sampling, other allowing precise sampling),
- LM survives on surface after ERV departure,
- Inbound single stage return transfer,
- ERC re-entry in Siberia.

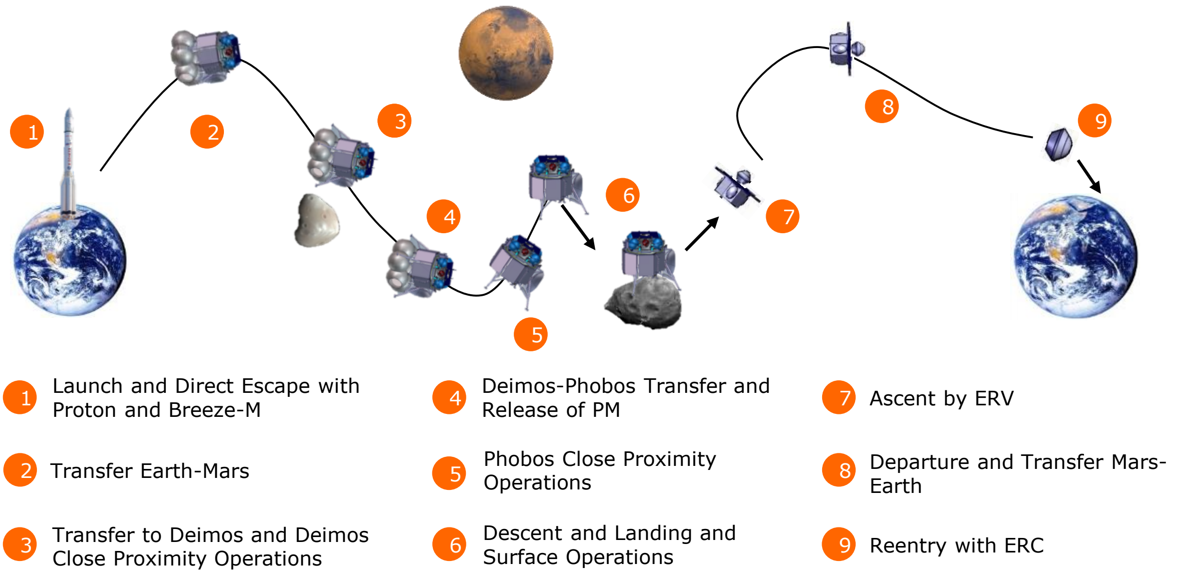


Figure 26-1: Reference Mission Architecture

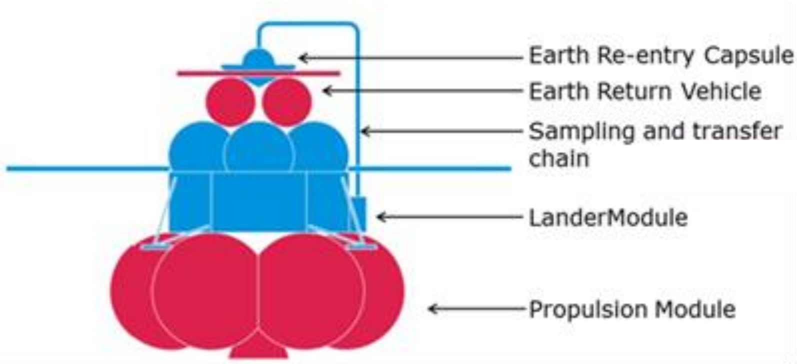


Figure 26-2: Baseline mission elements sharing (ESA = blue, RUS = red)

This has resulted into a design with the following characteristics:

Composite Main Characteristics	
Mass (inc. Margin)	Dry Mass: 1694 kg
	Science Instruments Mass: 38.4 kg
	Max Propellant Mass: 3377 kg (launch 2026)
S/C Main Components	- ERC (Earth Re-entry Capsule) - ERV (Earth Return Vehicle) - LM incl. science P/L (Lander Module) - PM (Propulsion Module)

Table 26-1: Composite main characteristics

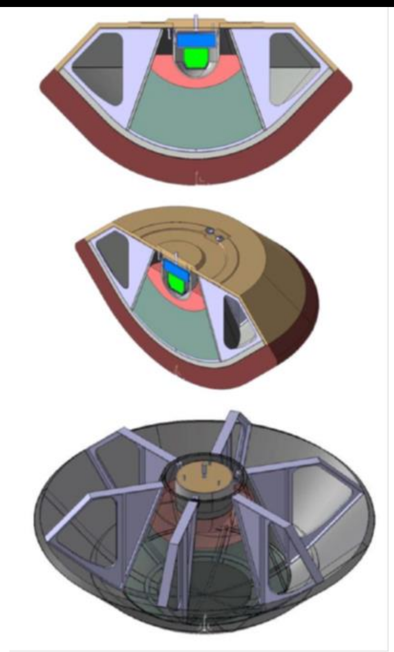
Earth Re-entry Capsule Description			
Trajectory	Landing location	Kazakhstan	
	Entry velocity	12.3 km/s (relative entry velocity - worst case retrograde)	
	FPA	-9.8 deg (nominal)	
	Mass	35 kg (incl. margin)	
Shape	Scaled from Hayabusa 45° half cone front shield		
	Main Diameter	0.75 m	
TPS	FS: ASTERM BS: Norcoat Liege		
	Heat load	Max: ¬ 221 MJ/m² (w. margin)	
	Heat Flux	Max: ¬ 15 MW/m² (w. margin)	
EDLS	None (no parachute)		
Structure	Load bearing Crushable materials to limit loads on sample		
Mechanisms	Sample container		
	Spin Separation device remaining on ERC		
GNC	None (uncontrolled re-entry)		
Communications	High g-load resistant recovery beacon based on aviation ELT or alternative		
DHS	None		

Table 26-2: Earth Re-entry Capsule main characteristics

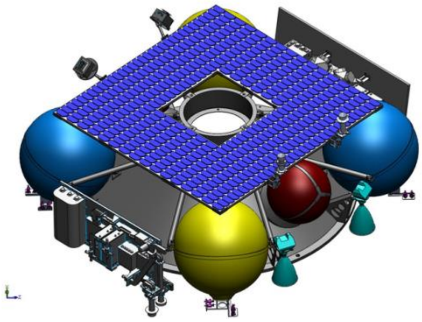
Earth Return Vehicle Characteristics			
AOCS/GNC	Sensors	Star Trackers	
		Sun Sensors	
		IMU	
	RCS	16 x 0.8N Thrusters, cold gas	
Propulsion	Bipropellant system, NTO/N2H4		
	Main Engine: 4 x 123.5N		
	Tanks: 4 fuel + 2 pressurant		
Power	SA	Body mounted	
	Battery	1 x Lithium Ion	
		BoL energy: 616 Wh	
	On-board voltage	27±1.35V	
Communications	All X-Band system		
	2 omni-directional antennas		
Thermal	MLI, heating lines, heaters		
DHS	OBC		
Mechanism	ERC spin separation device (SED TRP)		
	ERV separation remaining on ERV		
	Cable cutters		
	ERC ring hinge		
	ERC hold down		
Structure	Structural tanks + central cone for ERC accommodation		

Table 26-3: Earth Return Vehicle main characteristics

Lander Characteristics			
AOCS	D&L	Autonomous relative navigation	
	Sensors	2 x Star Tracker (AASTR)	
		2 x European IMU (Astrix 1090 + QA3000)	
		Wide Angle Cameras (2 OH) + (1EU), FoV: 53°	
		2 x Coarse Sun Sensor (TNO)	
		2 x Radar Altimeter	
	Actuator	4 x Reaction wheels (RSI 12/75-60)	
		16 / 24 x 20N thrusters	
Propulsion	Monopropellant system (Hydrazine)		
	Main engine: 1 x 1.1kN HTAE		
	Tanks	4 x Eurostar 2000 based, with 1801kg propellant	
Power	SA	5 x deployable wings	
		Solar cells: 30% 3J GaAs	
		Total area: 10.8 m²	
		1.2 kW (EOL Mars Orbit)	
	Battery	1 x Lithium Ion	
		BoL energy: 2600 Wh	
	Bus	28V MPPT regulated bus	
Communications	All X-Band system		
	1 x steerable HGA		
	3 x fixed LGA for 4π coverage		

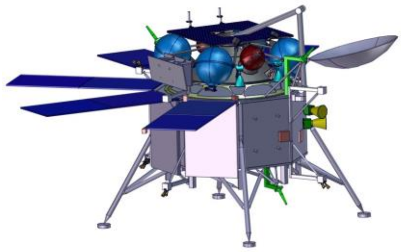
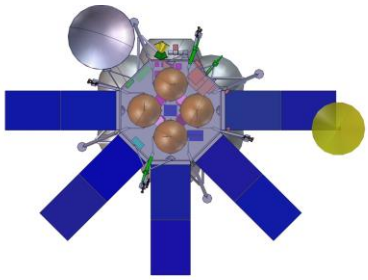
Lander Characteristics			
	2 x TWT Power: 65W		 
	2 x optional LGAs on PM		
Thermal	MLI, heating lines, Black Paint, SSM, insulating Stand-Offs		
	No heat pipes		
Mechanism	Sample	1 x Robotic arm incl. gripper	
		4 x landing legs	
		Sampling and containment tool (Rotary brushes)	
	Support	SA HDRM	
		HGA pointing mechanism	
		HGA pointing electronics	
		HGA resettable HDRM (RUAG)	
		Robotic arm HDRM	
		ERV separation device	
		ERV ejection springs	
DHS	OBC + MM based on LEON-FT		
Structure	Octagonal structure with CFRP and Al-Al panels. Corner beams transferring the load from the 8 PM hard points; top and bottom covers		

Table 26-4: Landing Module main characteristics

Propulsion Module Characteristics		
AOCS	None (Controlled by LM)	
Propulsion	Bipropellant system N2H2/NTO	
	Main engine: 20 kN	
	Tanks: 6 spherical	
Power	Chemical battery for propulsion power supply	
Communications	X-band antenna (Optional control by LM)	
Thermal	MLI, heaters	
DHS	None	
Structure	Structural tanks	

Table 26-5: Propulsion Module main characteristics

Furthermore, a backup scenario has also been studied (at a lower level compared to the baseline) following the same mission architecture than the baseline but with a different mission elements sharing as shown in Figure 26-3, its main characteristics are as follows:

- ERC (ESA) and ERV (RU) identical to the baseline mission,
- Switch of responsibility between ESA and Russia on the PM and LM resulting on a different staging: Jettisoning of the PM at Deimos arrival (wrt. Phobos arrival in the baseline) and Deimos to Phobos transfer by the lander propulsion system,
- Russian LM extensively based on Phobos-Grunt re-use,

- ESA PM largely based on the industrial Phootprint design heritage (TAS-I design).

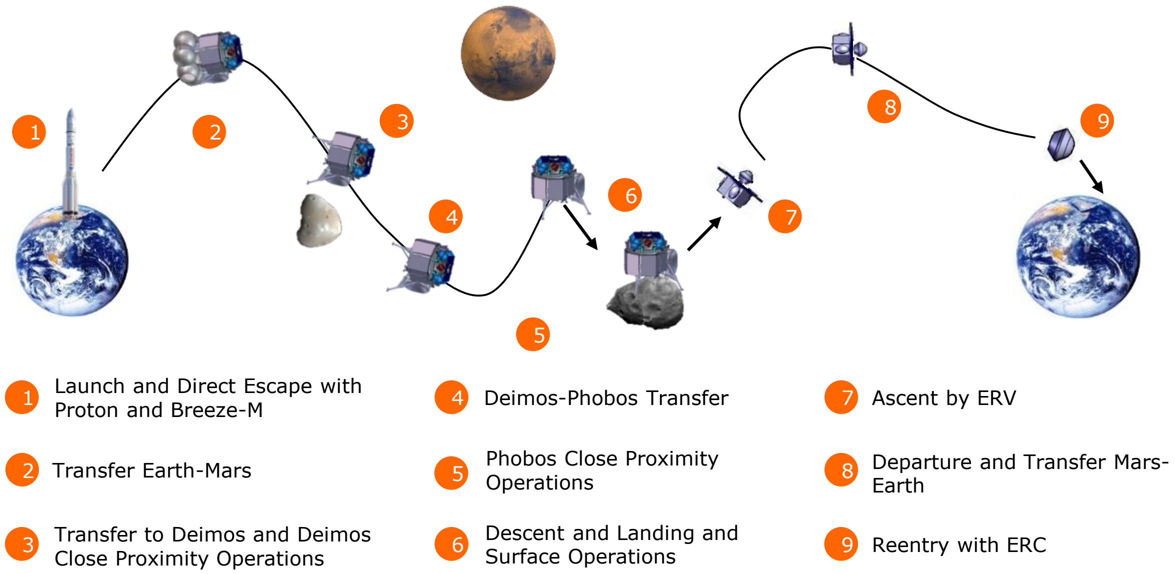


Table 26-6: Back-up Mission Architecture

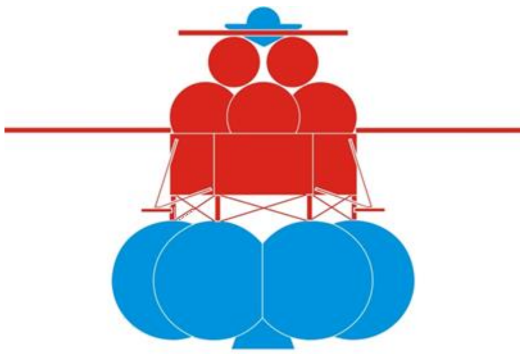


Figure 26-3: Backup mission elements sharing (ESA = blue, RUS = red)

26.2 Main Study Outcomes

The main outcomes of the study can be briefly summarised as follows:

- Mission Margin
  In the case of both baseline and backup scenarios, not only the mission observes and samples Phobos (primary mission objective) but it also extensively observes the surface of Deimos (secondary objective). This can be seen as both a Δv (ie. mass) and timeline margin since this phase can be either reduced (mostly then a time margin) or removed (allowing then saving around 1000 ms^{-1} Δv). Note that an intermediary option could be to limit this Deimos phase to only fly-bys (this would allow a significant Δv reduction but probably not so much time).

- Russian PM
  This PM has an important dry mass leading to an optimised staging when jettisoned as early as possible (eg. After MOI), in the case when the LM would use a bi-propellant propulsion system. However, as its use in this mission has been baselined, it can be used as a way to reduce the LM mass as much as possible (and therefore simplifying it) by transferring to it as much mission Δv as possible. Moreover it allows to use a hydrazine-based propulsion system for the LM, which simplifies the design. In this frame, the study has concluded that all outbound Δv up to Phobos arrival is performed by the PM.

- European LM
  - A design targeting a LM as light as possible as this one (see above) should implement a propulsion system as performing as possible such as bipropellant. However, for system simplicity reasons as well as aspects such as surface contamination during landing, a monopropellant system has been retained. This is possible due to the fact that no big manoeuvre is performed by the LM (only RCS are used), and the delta V is also not that high, so that the difference in mass between the 2 solutions is acceptable.
  - As for most Phobos missions, the latitude of the landing site is critical to the power subsystem design when implementing a mission using fixed SA. This is not only due to the difficulty of closing a balanced energy budget but also in aspects such as the shadowing of SA when landing in high latitudes.

- Interface between elements
  - PM-LM
    The attitude control system of the PM being only based on TVC of the main engine (Fregat heritage) has a non-negligible impact on the "lightweight" LM design in terms of DHS control and roll control of the stack by the PM
    In a power standpoint, the PM is self-sufficient as far as the TVC operation is concerned; however, power from the LM is required for thermal control. The current requirement is rather high (in the range of 300W permanent) and has a non-negligible impact on the overall LM power budget.
    At the exception of the previous, the interface with the PM can be seen as similar as when interfacing with a Fregat upper stage (which is the heritage of the PM)

  - LM-ERV
    The interface is rather straight forward and should not present any major challenge.
    One of the only major aspects to be kept in mind is the electrical interface between the two stages. Despite the independence of the ERV, power might need to be provided from the LM. Globally, this is not an issue, however this need should be minimised and be rather punctual (i.e. for

well defined and not too long mission phases), so that it does not impact to much the LM EPS sizing.

- ERV-ERC
  This interface can be seen in three aspects: Accommodation, attitude at separation (eg. Spin rate), and access for sample transfer. The study has concluded that based on the ERV current design and considering the constraints of the ERC (eg. Fragility of the ERC TPS material), all of the above do not seem to present any significant challenge.

26.3 Further Study Areas

In order to complete the design achieved during this study, it is suggested to further study the following areas:

- Science requirements
  - Consolidation of the requirements related to the samples in order to allow a more precise sampling tool design (eg. Depth, individual sampling ...).
  - Definition of requirements related to the observation of Deimos to better define the Deimos observation phase.
  - Consolidation of the landing accuracy requirement
  - Consolidation of the resolution required for global and local mapping
  - Consolidation of the soil contamination requirement (from the thrusters)
  - Consolidation of the sample g-load requirement at landing on Earth
- Technical requirements: consolidation of ESA/ROSCOSMOS interfaces (e.g. for power and communications)

26.4 Final Considerations

This study has been the first CDF study where ESA has collaborated in real time with its Russian partners IKI and Lavochkin.

Beyond demonstrating the technical feasibility of a combined Russian/European Phobos sample return mission, this study has proven that real-time concurrent activities can successfully be performed between ESA, IKI and Lavochkin. Repeating this experience in the future can therefore only be encouraged.

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27 REFERENCES

RD[1] Phobos Sample Return Roscosmos-ESA joint mission Mission Requirement Document , CDF-PSR-MRD issue 1, 15/04/2014

RD[2] M. Khan: Phootprint Mission Analysis Guidelines (MAG) Issue 2 Rev. 0, SRE-FP/MREP/PHO-MAG, ESA/ESOC 2013/8/14

RD[3] Proton Mission Planner's Guide, Revision 7, ILS, July 2009

RD[4] Phootprint – Phobos Sample return Mission Science Requirements Document (SciRD),PHO-RSSD-RS-001, issue 1, revision 0, 24 April 2013.

RD[5] Ishii, N., Hiraki, K. and Inatani, Y.: Attitude Motion and. Aerodynamic Characteristics of MUSES-C Re-entry Capsule,. ISAS Report SP No. 17, 2003

RD[6] Program TRAJ3D V2.3 three degree of freedom planetary entry and descent trajectory simulation user manual. Fluid Gravity Engineering Ltd.

RD[7] Earth Global Reference Atmospheric Model (Earth-GRAM 2010)

RD[8] U.S. Standard Atmosphere, 1976, U.S. Government Printing Office, Washington, DC, 1976

RD[9] Detra RW, Hidalgo H. Generalised heat transfer formulas and graphs for nose cone re-entry into the atmosphere. ARS J 1961.

RD[10] Tauber, M. E.; Sutton, K. Stagnation-point radiative heating relations for earth and Mars entries, Journal of Spacecraft and Rockets, 28, 40-42

RD[11] Andersen, Hypersonic and high temperature gas dynamic

RD[12] H. Julian Allen H. J., A.J. Eggers A.,J. : A Study of the Motion and Aerodynamic Heating of Ballistic Missiles Entering the Earth's Atmosphere at High Supersonic Speeds, NACA Report 1381, 1958

RD[13] Proton Launch System, Mission Planner's Guide, ILS International Launch Services, Revision 7, July 2009, www.ilslaunch.com.

RD[14] Preliminary Design Document and Scalability Analysis, Study report, B-U. Halter, 2.1, 28/06/2013

RD[15] Composite Design Report, Study report, Phootprint Team, 1.0, 2014

RD[16] MMSR_A5, Moon Mars sample return, Study report, CDF TEAM, 1.0, 2012

RD[17] Margin Philosophy For Science Assessment Studies, A. Atzei, A. Lyngvi, A. Wielders Issue 1, Rev3, 15/06/12

RD[18] "Mission Analysis and Trajectory GNC for Phobos Proximity Phase of Phootprint Missison" IAA-AAS-DyCoSS2-12-04, F. Cacciatore and J. Martin. http://www.dycoss.com/program/final/IAA-AAS-DyCoSS2-14-12-04.pdf

RD[19] "Orbital Operations for Phobos and Deimos Exploration" Mark S. Wallace et al. AIAA paper.

RD[20] "Phootprint Pre-Phase A System Study. Technical Note 04. (ENG04). Composite Design Report. Issue 1" PSR. ASU.TN.003. Airbus Defence & Space.

RD[21] "Phootprint Pre-Phase A System Study. ENG02: Phootprint Mission Analysis and Operations. Issue 2" TASI-SD-PH-ARP-0001. Thales Alenia Space.

RD[22] "Phootprint Pre-Phase A System Study. ENG04: Phootprint Composite Design Report. Issue 1" TASI-SD-PH-DDD-0001. Thales Alenia Space.

RD[23] CDF Study Report CDF-131(A), Moons of Mars Sample Return using an Ariane-5 launcher (MMSR-A5)

RD[24] Space Mission Analysis and Design, James R. Wertz, Space Technology Library, 3rd edition.

RD[25] http://www.russianspaceweb.com/fregat.html

RD[26] "SpaceLab: A spacecraft Simulation Laboratory under Matlab/Simulink (version 2.0) Sept 2006", Shufan Wu and Finn Ankersen.

RD[27] Assessment and Breadboarding of a Planetary Altimeter (ABPA), Statement of Work, Contract number AO/1-6711/11/NL/MH, issue 2, 06/12/2010

RD[28] Statement of Work "Mars Robotic Exploration Preparation-2 Programme. Guidance, Navigation and Control (GNC) for Phootprint descent and landing" Programme Reference: E905-017EC.

RD[29] Marco Polo Capsule Retrieval Assumptions Document, Rainer Timm, DOPS-MGT-TN-1002-OPS-HSA, issue 1, revision 1, 23/07/2009

RD[30] Technology Readiness Levels Handbook for Space Applications, TEC-SHS/5551/MG/ap, Issue 1 Revision 6, Dated September 2008

RD[31] CDF Study Report "Moons of Mars Sample Return using an Ariane 5 Launcher" CDF-131(A), May 2012

RD[32] Space Engineering Verification Guidelines, ECSS-E-HB-10-02A, 17-12-2010

RD[33] Call for Proposals to foster and promote Swiss scientific and technological competences related to space activities Technology Development, Swiss Space Office, Ref: 2012/DOC/002, Annex B Technological readiness levels (TRL), http://space.epfl.ch/webdav/site/space/shared/industry_media/Annex_B.pdf

RD[34] ECSS Standard Space Engineering Verification, ECSS-E-ST-10-02C, 6 March 2009

RD[35] ECSS Standard Space Engineering Testing, ECSS-E-ST-10-03C, 1 Juni 2012

RD[36] Space Project Management, Risk management, ECSS-M-ST-80C, 31 July 2008.

RD[37] Soyuz user's manual issue 2 revision 0 March 2012

28 ACRONYMS

Acronym	Definition
A/D	Analogue/Digital
ACS	Attitude Control System
AD	Applicable Document
AIT	Assembly, Integration and Test
AIV	Assembly, Integration and Verification
AOCS	Attitude Orbit Control System
APM	Antenna Pointing Mechanism
AS	Adapter System
AVM	Avionics Verification Model
BCDR	Battery Charge/Discharge Regulator
BoL	Beginning of Life
BoM	Beginning of Mission
C&DH	Command and Data Handling
CAD	Computer-Aided Design
CCD	Charge Coupled Device
CDF	Concurrent Design Facility
CFRP	Carbon Fibre Reinforced Plastic
CLUPI	Close Up Imager
CoG	Centre of Gravity
CoM	Centre of Mass
COTS	Commercial Off The Shelf
CU	Control Unit
D&L	Descent and Landing
DDTC	Department of Defense Trade Controls
DELIAN	Dextrous Lightweight Arm for Exploration
DHS	Data Handling System
DoD	Depth of Discharge
DoF	Degree of Freedom
DSA	Deep Space Antenna

Acronym	Definition
DSM	Deep Space Manoeuvre
EBB	Elegant Breadboard
ECSS	European Cooperation on Space Standardisation
EEE	Electronic, Electrical and Electromechanical
EGSE	Electrical Ground Support Equipment
EIRP	Equivalent Isotropic Radiated Power
EM	Engineering Model
EMC	Electro-Magnetic Compatability
EoCV	End of Charge Voltage
EoL	End of Life
EPS	Electrical Power Subsystem
EQM	Engineering Qualification Model
ERC	Earth Re-entry Capsule
ERV	Earth Return Vehicle
ESB	Earth Swing-by
FCL	Foldback Current Limiter
FCT	Flight Control Team
FDIR	Failure Detection Isolation and Recovery
FEM	Finite Element Model
FMHF	Free Molecular Heat Flux
FoV	Field of View
FPA	Flight Path Angle
FR	Final Review
GL	Gravity Loss
GMM	Geometric mathematical model
GNC	Guidance Navigation and Control
GS	Ground Segment
GSE	Ground Support Equipment
HDRM	Hold Down and Release Mechanism
HGA	High Gain Antenna
HKTM	HouseKeeping TeleMetry

Acronym	Definition
HTAE	High Thrust Apogee Engine
I/O	Input/Output
IKI	Space Research Institute of Russian Academy of Science
Imp	Maximum Power Point Current (Solar Cell)
IMU	Inertial Measurement Unit
Isc	Short Circuit Current (Solar Cell)
ITAR	International Traffic in Arms Regulation
ITT	Invitation to Tender
LCL	Latched Current Limiter
LGA	Low Gain Antenna
LEO	Low Earth Orbit
LEOP	Launch and Early Operations Phase
LGA	Low Gain Antenna
LM	Lander Module
LP	Launch Period
LPC	Launch Period Close
LPO	Launch Period Open
LuM	Lunar Mission
LV	Launch Vehicle
MAG	Mission Analysis Guidelines
MFLOPS	Million Floating-Point Operations per Second
MGA	Medium Gain Antenna
MIB	Minimum Impulse Bit
MidIR	Mid Infrared Spectrometer
MIPS	Million Instructions per Second
MLI	Multi layer insulation
MM	Main Manipulator
MMSR	Moons of Mars Sample Return
MOC	Mission Operations Centre
MOI	Mars Orbit Insertion
MPPT	Maximum Power Point Tracker (Tracking)

Acronym	Definition
MRD	Mission Requirements Document
MREP	Mars Robotic Exploration Preparation
MSR	Mars Sample Return
MSSR	Moons of Mars Sample and Return
NAC	Narrow Angle Camera
OBC	On-Board Computer
OCS	Onboard Control Complex
PAS	Payload Adapter System
PCDU	Power Conditioning and Distribution Unit
PCU	Power Conditioning Unit
PDU	Power Distribution Unit
PFM	Protoflight Model
PhSM	Phobos Soil Mission
PhSR	Phobos Sample Return
PICA	Phenolic Impregnated Carbon Ablator
PLF	Payload Fairing
PM	Propulsion Module
PP	Planetary Protection
PSM	Payload System Mass
PVA	Photovoltaic Assembly
QM	Qualification Model
QSO	Quasi Satellite Orbit
RA	Robotic Arm
RD	Reference Document
RF	Radio Frequency
RGB	Red, Green And Blue
S/C	Spacecraft
S3R	Sequential Switching Shunt Regulator
SA	Solar Array
SA	Solar Array
SAA	Solar Aspect Angle

Acronym	Definition
SADM	Solar Array Drive Mechanism
SAR	Solar Array Regulator
SATC	Sample Acquisition Transfer and Containment System
SC	Sample Container
SC	Spacecraft
SD	Sampling Device
SED	Spin and Ejection Device
SEU	Single Event Upset
SM	Structure Model
SoC	State of Charge
SOC	Science Operations Centre
SPARC	Scalable Processor Architecture
SpC	Spacecraft
SPF	Single Point Failure
SRE	Science Robotics & Exploration (ESA Directorate)
SRF	Sample Receiving Facility
SS	Summer solstice
SSM	Second surface mirror
Std.	Standard
StereoCam	Stereo Camera
STM	Structural Thermal Model
TBC	To Be Confirmed
TBD	To Be Decided
TC	Telecommand
TCM	Trajectory Correction Manoeuvre
TCS	Thermal Control System
TDA	Technology Development Activity
TDP	Technology Development Plan
TM	Telemetry
TOA	Target Orbit Acquisition
TPS	Thermal Protection System

Acronym	Definition
TRL	Technology Readiness Level
TT	Transportation Tool
TT&C	Tracking, Telemetry and Command
TWTA	Travelling Wave Tube Amplifier
US	Upper Stage
USO	Ultra Stable Oscillator
VIS/NIR	Visible/Near Infrared
VisNIR	Visible and Near Infrared Spectrometer
Vmp	Maximum Power Point Voltage (Solar Cell)
Voc	Open Circuit Voltage (Solar Cell)
WAC	Wide Angle Camera
WS	Winter solstice